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NAS9-150

31 January 1963

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FOREWORD

The Quarterly Reliability Status Report is submitted in accordance with the Apollo documentation requirements delineated in NASA Contract NAS9-150, Paragraph 4.5.4.7, of "Project Apollo Spacecraft Development Statement of Work," Part 4, dated 18 December 1961, and MIL-R-27542, Paragraph 5.4.3. The information contained herein covers the period from 1 October through 31 December 1962.

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SUMMARY

The reliability analyses performed during the past quarter indicate that the mission success reliability requirement of 96.5 per cent can be met by proper subsystems design with a reasonable number of on-board spares for the electronic subsystems. Based on these analyses, a new reliability apportionment for the lunar orbital rendezvous (LOR) mission, with a list of the necessary subsystem apportionments to meet the mission success requirement of 0.90, has been made. Table 1 lists the subsystem mission success requirements for the LOR mission and the previous subsystem mission success requirements apportioned for the direct landing mission and indicates those subsystems affected by its redirection to the LOR mission. The higher subsystem requirements in the LOR mission result from the change of the spacecraft requirement of 96 per cent in the direct landing mission to 96.5 per cent in the LOR mission.

The basic analysis establishing the subsystem requirements for the LOR mission was made using high-reliability parts for the mechanical and electrical subsystems; high-reliability parts for the communication and data, stabilization and control, and instrumentation subsystems; and Massachusetts Institute of Technology estimates for the guidance and navigation subsystem. A second analysis was made using high-reliability parts in the guidance and navigation subsystem to evaluate the reduction in spares weight that would result from the use of these parts. An additional analysis was made, using present state-of-the-art parts, to resolve the reliability status of the subsystems and to determine the areas where future reliability effort can best be spent.

SYSTEM AND SUBSYSTEM TEST ACTIVITY

Spacecraft Tests

Qualification test requirements and test criteria were provided for operational test procedures for boilerplate 22, high-altitude abort; boilerplate 23, backup for boilerplates 6 and 12; airframe 001, propulsion spacecraft; airframe 006, vibration-acoustic tests; and airframe 008, environmental proof spacecraft. This information included the performance and environmental parameters during flight that will be analyzed by Reliability. No special tests were imposed on boilerplates 9, 15, 16, and 18 since their tests are for qualification and demonstration of launch vehicle and spacecraft compatibility only.

[redacted]

Table 1. Apollo Lunar Orbital Rendezvous Mission, Apportioned Reliability (Polaris Components for Guidance and Navigation)

Subsystem	Apportioned Lunar Orbital Rendezvous	Apportioned Direct Flight
Structural and mechanical	0.999926	0.997
Launch escape system	0.999989	0.999989
Electric power system	0.9989	0.9955
Earth land	0.99994	0.99994
Cryogenic storage	0.9989	0.9989
Service module reaction control system	0.999409	0.996
Command module reaction control system	0.999969	0.99996
Lunar excursion module	0.984	
Environmental control subsystem	0.997675	0.9935
Service propulsion system	0.99977	0.99977
Boosters	0.95	0.95
Stabilization and control Guidance and navigation Communications and data Instrumentation	0.994558 0.988901 0.987735 0.998502	0.995 0.994 0.9925 0.9955

Reliability facility requirements (test equipment, space, and office equipment) have been established for a field test site at the White Sands Missile Range (WSMR) and the Atlantic Missile Range (AMR). To reduce reaction time, provisions have been made for local analysis of failures.

Four drop tests were conducted with boilerplate 1. These were water impact tests monitored by Reliability for the purpose of developing environmental test criteria.

A qualification and development test schedule was devised to determine the type of testing to be performed on each major component and subsystem used in boilerplate and airframe articles and the type of environment to which they will be subjected. A record will be kept in this test schedule of the number and date of all qualification and development tests made, the type of test, and the extent of testing experienced by any system component prior to a flight.

Integrated Systems Testing

A status review of the Apollo equipment qualification in a representative environment prior to the first manned flight has been made by a task force that was organized to define this problem area and recommend a definitive program. To define the problem, the approach has been to establish system constraints on all subsystems that will be used in order to determine whether an article is qualified and/or ready to launch for each successive flight. The constraints will be established as a joint effort between NASA and S&ID. From this, specific criteria for qualification and launch can be established, and, where necessary, corrective or compensatory action can be determined at any point in time.

Tentative constraints were set for all subsystems to be employed as an integrated system on the first flight article, boilerplate 6. These constraints were reviewed with NASA, Houston, Texas, during 17 to 20 December 1962. The results and direction of that meeting will be used to develop the subsystem constraints for each succeeding flight article. The intent of this effort is to establish and graphically present the fundamental information necessary for the decision to fly or to schedule postponement. From this effort, point-by-point qualification goals for first manned flight may be reviewed periodically by subsystem and program adjustments made as required. This effort will constitute a portion of the Multiple System Tests, SID 62-109-5.

Subsystem Test Activity

Ground rules were established to accomplish the objectives of the reoriented plan and were applied to each subsystem according to its

respective contribution toward mission success and crew survival. Mission profiles, dynamic test levels, criticality indices, and minimum test criteria were formulated for individual systems on the basis of application considerations. The results of this effort were tabulated as changes to the originally proposed Qualification-Reliability Test Plan and submitted as part of the Big Four Delta of the reoriented program. A summary of the effected changes to each subsystem is reported by individual system in the following paragraphs of this section of the report. It should be noted that reliability assessment will be accomplished without additional hardware being required beyond the qualification testing phase; however, any failures encountered will impose a requirement for additional tests and hardware. In addition to the changes of the reoriented program, the test requirements and effort necessary for the effective integration of the lunar excursion module into the Apollo spacecraft configuration and mission objectives under the reoriented plan of qualification were developed and presented by individual subsystem.

INTRODUCTION

This section presents a mission success reliability analysis of the Apollo spacecraft, defines areas where discrepancies exist between the required and estimated reliability levels, and defines plans for corrective action.

Estimates were made utilizing state-of-the-art equipment failure rates and high-reliability part failure rates. A Monte Carlo program employing IBM 7090 equipment was used for predicting spacecraft reliability. This report contains a description of the Monte Carlo program and a justification for its use.

A time-line analysis defining the various functions that must be performed throughout the mission is included. Each system is described with an explanation of the use of the components within the subsystem. Individual equipment reliability estimates are presented. The general abort criterion described, and the specific equipments within each subsystem that would cause an abort are listed, and the results of these analyses are presented and discussed for both high-reliability and state-of-the-art parts. A list of the on-board spares required for the electronic subsystems to meet the reliability requirements is included.

During the next period, this analysis will be expanded to include crew safety calculations and mission success. Any additional system modifications necessary to meet the crew safety requirements will be discussed at that time.

METHOD OF ANALYSIS

The complexity of the Apollo spacecraft systems coupled with the complexity of the Apollo mission itself dictates the necessity of a reliability analysis model that (1) will relate component reliability to spacecraft reliability, (2) may be used to apportion the required component reliability, (3) will facilitate system trade-off studies, and (4) can provide a means of assessing the status of both the mission and crew safety reliabilities. The requirements for this model are as follows:

1. Simple program logic
2. Minimum input data
3. Fast reaction to changes
4. Maximum output data per calculation

After thorough investigation, three programs were considered, logic equations, combinations of successes and failures, and Monte Carlo technique using the IBM 7090.

LOGIC EQUATIONS

A typical logic diagram for the Apollo spacecraft is presented in the upper left-hand portion of Figure 1. F-1 and F-2 in these diagrams show the equipment that is required to perform various functions for the various phases of the mission. Certain features of these diagrams make it extremely difficult, if not impossible, to calculate reliability by conventional methods. One feature is the use of component C for two functions. A typical example is the body-mounted gyros component that has a primary function in the stabilization and control system, but is used as an alternate mode in the navigation and guidance system to back up the inertial measuring unit. A second feature is that a particular component, such as component C, may function in series for one phase of the mission and function in parallel or as a redundant component for the second phase of the mission.

A third feature that complicates the problem is the redundancy within the system for the various phases. This redundancy makes the probability expressions for phase 1 and phase 2 mathematically dependent; hence, these expressions cannot be multiplied.

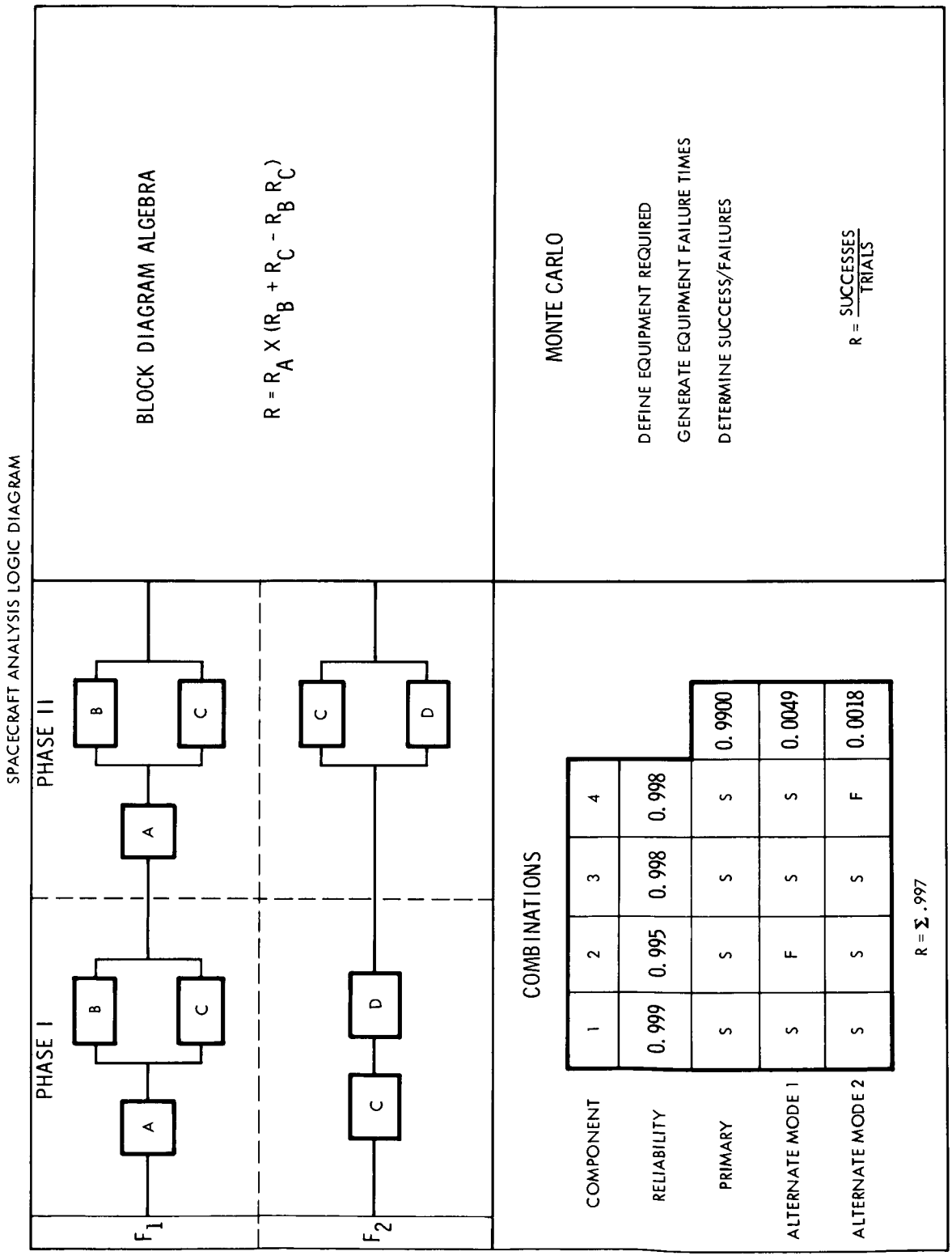

Figure 1. Spacecraft Analysis Sample Logic Diagram

In evaluating the block diagram algebra approach to reliability calculations as illustrated in the top right-hand corner of Figure 1, it can be seen that expressions cannot be written for the two phases and simply multiplied together because of the aforementioned dependency. Furthermore, this approach of multiplying the reliability expressions for the various functions cannot be used because certain components appear in two functions, thus making these expressions mathematically dependent. Hence, the normal approach of using block diagram algebra is virtually impossible to use for the Apollo system.

COMBINATIONS OF SUCCESSES AND FAILURES

The second program considered was that of appraising all combinations of successes and failures for the various phases of the mission, determining the combinations that allow successful completion of the mission, and calculating the probability of one of these combinations occurring. As shown in the lower left-hand corner of Figure 1, this is done by determining the probability of each one of these combinations occurring and then summing these probabilities to determine spacecraft reliability. This approach precludes the problem of dependency between functions, but is very time consuming even for one phase of the mission. The major problem, however, arises in combining the expressions for the various phases of the mission. To calculate the reliability for a particular phase, it is necessary to consider all combinations of equipment with which the previous phase could have been completed. As can be seen in the chart in the lower left-hand corner of Figure 1, it would be necessary to consider the probability of completing that phase in the primary mode (i. e., with everything operating 0.999) and multiplying this by the probability of at least one of the alternate modes that are required in the following phase is operating. To this would be added the product of the probability of completing the first phase in the first alternate, 0.0049, by the probability of completing the following phase with that equipment that was operating the first phase, considering all the alternate modes for the second phase. This process must be continued for all the modes in which the first phase can be completed. The same process must be continued for all phases of the mission. The number of calculations pyramids rapidly as the process is continued through the mission, and the number of calculations increases to an astronomical number. The number of calculations that would have to be performed becomes so high that this method becomes completely impractical, if not impossible.

MONTE CARLO TECHNIQUE

The third method considered, and the one which was decided upon, is the Monte Carlo or mission simulation technique. This process consists of

generating, with a random number generator, the number of failures that would occur in a particular mission and the time at which they would occur. Logic diagrams are constructed to define those combinations of equipment that are necessary to complete each function for each phase of the mission. The failures are then compared to the logic diagrams to determine whether or not a mission can be completed with that combination of failures occurring at the times determined. This process is repeated for a large number of missions and the percentage of completed missions per attempted mission is calculated. The major advantage to this approach is that the logic diagrams need not be mathematically independent as in the other two methods. All configuration data are fed into the program as input data. Hence, when configuration changes are made, it is not necessary to modify the basic program logic but only to modify the input data. The Monte Carlo program, thereby provides a method that makes it possible to calculate spacecraft reliability with a reasonable expenditure of manpower and provides a method that is very flexible.

The procedure evaluation chart, Figure 2, summarizes the evaluation of the methods of reliability analysis considered. Figure 2 clearly favors the Monte Carlo technique, which has as an additional merit the versatility of the finished program. Since both the mission and the hardware logic are compiled by the computer from the input data, any form of a mission desired may be described, and the program immediately becomes applicable.

INPUT DATA

Two types of input data are required for the reliability model. One pertains to the system relationships, and the other pertains to the mission description.

The system data describe the physical relationships of the system, such as the number of components, failure rates of components, and relationship between components (i.e., alternate, primary or dependent), and the functions the components support. These are originally described by block diagrams, typical examples of which are shown in a later section of this report. The mission data describe the time sequence of functions that the components are required to support and the relations of the functions to mission success.

Figure 3 depicts the steps taken in the computer simulation. The first step of the program is to generate, through the use of a random number generator, operating times to failure for each component based on its failure rate. These times to failure are then compared with the mission operating time required of the component for the mission. If all the components have times to failure that are greater than those required of the

	LOGIC SIMPLICITY	INPUT SIMPLICITY	REACTION TO CHANGE	OUTPUT DATA/RUN
ALGEBRA	POOR	POOR	POOR	POOR
COMBINATIONS	POOR	POOR	FAIR	FAIR
MONTE CARLO	EXCELLENT	FAIR	EXCELLENT	EXCELLENT

Figure 2. Spacecraft Analysis Procedure Evaluation

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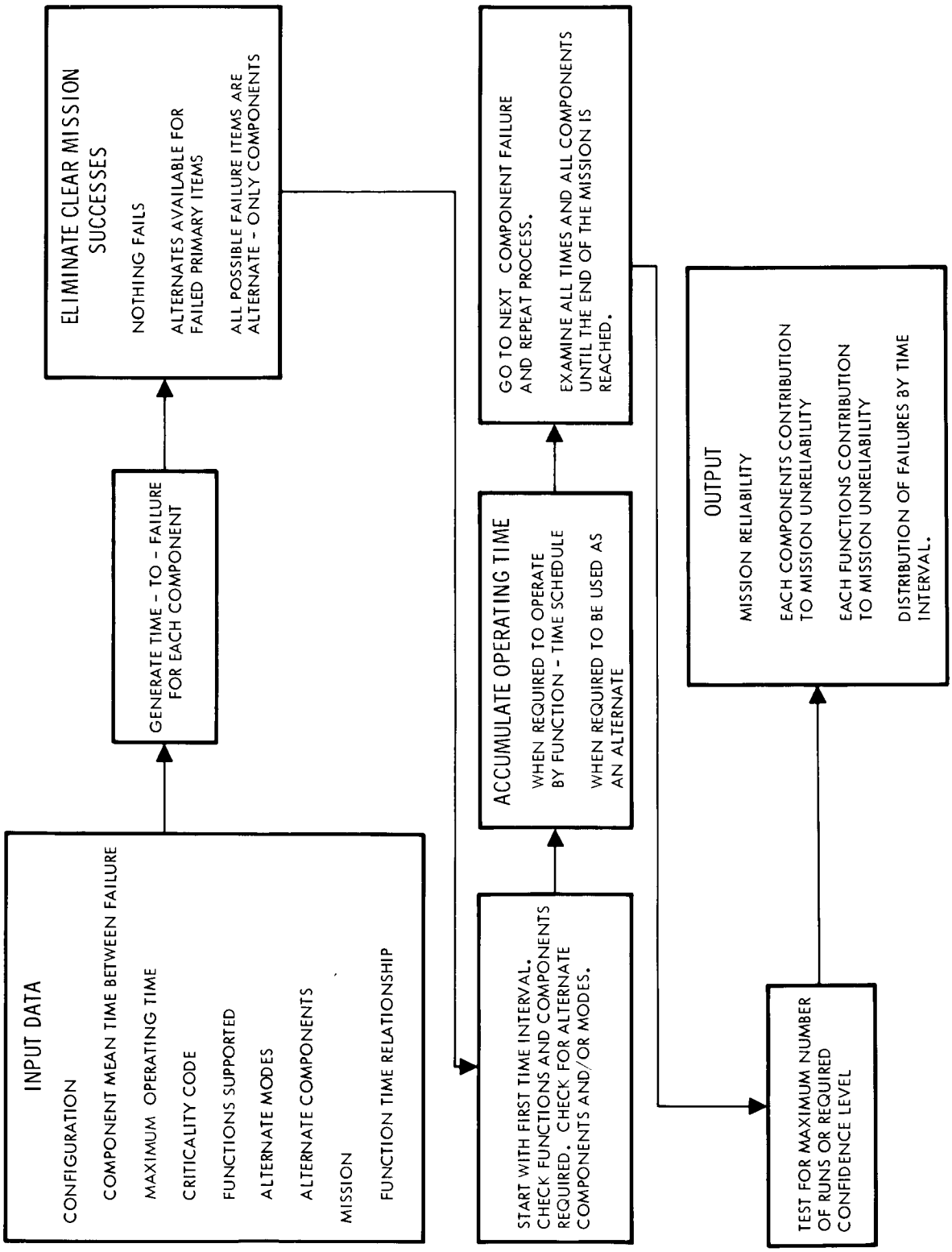
INPUT DATA
CONFIGURATION
COMPONENT MEAN TIME BETWEEN FAILURE
MAXIMUM OPERATING TIME
CRITICALITY CODE
FUNCTIONS SUPPORTED
ALTERNATE MODES
ALTERNATE COMPONENTS
MISSION
FUNCTION TIME RELATIONSHIP
GENERATE TIME - TO - FAILURE FOR EACH COMPONENT
ELIMINATE CLEAR MISSION SUCCESSES
NOTHING FAILS
ALTERNATES AVAILABLE FOR FAILED PRIMARY ITEMS
ALL POSSIBLE FAILURE ITEMS ARE ALTERNATE - ONLY COMPONENTS
START WITH FIRST TIME INTERVAL. CHECK FUNCTIONS AND COMPONENTS REQUIRED. CHECK FOR ALTERNATE COMPONENTS AND/OR MODES.
ACCUMULATE OPERATING TIME
WHEN REQUIRED TO OPERATE BY FUNCTION - TIME SCHEDULE
WHEN REQUIRED TO BE USED AS AN ALTERNATE
GO TO NEXT COMPONENT FAILURE AND REPEAT PROCESS.
EXAMINE ALL TIMES AND ALL COMPONENTS UNTIL THE END OF THE MISSION IS REACHED.
TEST FOR MAXIMUM NUMBER OF RUNS OR REQUIRED CONFIDENCE LEVEL
OUTPUT
MISSION RELIABILITY
EACH COMPONENTS CONTRIBUTION TO MISSION UNRELIABILITY
EACH FUNCTIONS CONTRIBUTION TO MISSION UNRELIABILITY
DISTRIBUTION OF FAILURES BY TIME INTERVAL.

Figure 3. Spacecraft Analysis Program Flow Chart

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components in the mission, the mission is logged as a success and a new mission is started. A mission failure is possible if some components have times to failure less than that required in the mission. These components are then divided into alternate components, primary components that have alternate components, and primary components that do not have alternate components.

A primary sifting operation is performed on these three groups to determine if a mission success is possible. The following conditions must be met to determine a mission success without mission simulation.

1. Alternate components have failed (Group 1) but are not required to function since their primary component has not failed.

2. Primary components have failed but their alternate components (Group 2) have not failed.

3. No primary components that have no alternate components (Group 3) have failed.

If the previously described conditions are not met, the components that may fail in the mission are then tested through the simulation of the mission. The components to be tested for mission simulation are primary components that fail in the mission and have no alternate components, primary components that fail in the mission whose alternate components may have failed, and those alternate components that fail in the mission and support primary components that have failed. The failure of these components may contribute to or cause a mission failure. The mission simulation begins with the examination of the first time interval. All components supporting functions that operate in this interval are tested to see if they have failed during this interval, i.e., if the accumulated operating time exceeds the operating time to failure generated for them. If there is no failure of these components, the time accumulated for each component is stored, and the next missioninterval is similarily examined. If a component fails, it is examined to see if it has an operating alternate component to support it. If not, the mode of operation in the function that it supports would be failed, and that function would be examined for the availability of an alternate mode of operation. In the event that neither an alternate component nor an alternate mode of operation within the function is available and the function is critical, a mission failure has occurred. At the point of mission failure, the mission simulation stops, and the components and functions causing this mission failure are stored in their appropriate arrays to be printed out when the required number of missions has been run.

The required number of missions is determined by input stipulation or through the computation of the confidence level. This confidence level, which is a function of missions attempted and mission failures, is also stipulated by input.

PROGRAM OUTPUT DATA

The following output data are acquired from this program:

1. Number of missions run
2. Number of successful missions
3. Number of failed missions
4. Observed reliability
5. Confidence level on observed reliability
6. The number of times mission failure was caused by the failure of each function
7. The number of missions that failed in each time interval

The output data acquired for each component are as follows:

1. Number of times component was involved in a mission failure
2. The measure of effectiveness of each component, which is number of missions run divided by number of times the component failed
3. Number of times component failure initiated mission failure
4. Number of times component failure terminated the mission

PROGRAM LIMITATIONS

The following are the limitations of the present model and are subject to improvement with the installation of the IBM 7094 expected by June 1963.

1. A maximum of 300 components
2. Each component may support a maximum of 10 functions

3. Each primary component may have a maximum of 12 alternate components

4. Each component may have 110 dependent components

5. A maximum of 250 functions

6. Each function may have 12 modes

7. There is a maximum of 59 time intervals

8. A maximum of 400 functions in the time function schedule

9. The maximum number of missions is 32,678

DEFINITIONS

The following definitions are pertinent to the program:

1. Component - Any device whose reliability can be expressed as $R = e^{-\lambda t}$, where λ is the failure rate (constant by definition within the limits of the intended mission) in terms of failures per hour, and t is the operating time in hours.

2. Primary component - A component that performs in the preferred mode of operation in a function

3. Alternate component - A component that replaces a failed component and it is characterized by either of the two types: standby, an alternate that is turned on only to replace a failed component; and full-time redundant, an alternate that operates whenever the component it backs up operates and which can act as a replacement in case of failure of the component it backs up

4. Dependent component - A component whose mode of operation is affected by the failure of another component and has two types: must fail, a component that must fail when the component on which it is dependent fails; and must not fail, a component that must not fail when the component on which it is dependent fails

5. Mission - A set of functions occurring in a specified time sequence and which may be broken into time intervals of nonuniform length in which various functions are required.

6. Function - The characteristic action of a component or group of components. In the course of a function components acquire

operating cycles and/or operating time according to the discrete operating intervals of a function. A function may be: critical if its failure results in mission failure or noncritical if its failure will not cause a mission failure.

MISSION PHASE TIME LINE

The lunar orbital rendezvous mission, which was used for the reliability analysis presented in this report, has been broken down into 59 phases as shown in Table 2. This breakdown was necessary because some of the equipment is operated for only specific times during the mission. Furthermore, the equipment required to perform functions during particular phases varies. This is particularly true with the navigation and guidance equipment, which is operated only when it is necessary to take navigational sightings or to align the spacecraft for application of delta velocities. At other times during the mission, the equipment is in operation but is only monitored. Typical examples of the separate logic block diagrams that were constructed for each of the 59 phases and are included in the Monte Carlo program are shown in the next section, Subsystem Analysis.

Table 2. Time-Line Lunar Orbital Rendezvous Mission Maximum Times Expanded for Monte Carlo Computer Study

Phase	Beginning of Phase Time	End of Phase Time	Phase Time (hr)	Phase	Beginning of Phase Time	End of Phase Time	Phase Time (hr)
1 Launch to 100, 000 feet	0.00	0.028	0.028	31 Command module navigational sightings	145.660	146.660	1.000
2 100, 000 feet to jettison of LET	0.028	0.042	0.014	32 LEM lunar surface stay period 2	146.660	162.160	15.500
3 LET jettison to earth orbit injection	0.042	0.280	0.158	33 Lunar ascent to circular orbit at 50, 000 feet	162.160	162.230	0.070
4 Earth orbit stay (3 orbits)	0.200	4.700	4.500	34 Ascent to 80 nautical miles	162.230	163.200	0.970
5 Translunar injection	4.700	4.780	0.080	35 Rendezvous and docking	163.200	170.900	7.700
6 Translunar coast cisearth period 1	4.780	4.947	0.167	36 Lunar orbit navigations	170.900	172.900	2.000
7 LEM transfer	4.947	5.447	0.500	37 Lunar orbit coast	172.900	196.230	23.330
8 First midcourse correction	5.447	7.947	2.500	38 Lunar orbit navigation sightings	196.230	197.230	1.000
9 Translunar coast period 2	7.947	25.947	18.000	39 Lunar orbit coast	197.230	221.570	24.340
10 Second midcourse correction	25.947	27.947	2.000	40 Lunar orbit navigation sightings	221.570	222.570	1.000
11 Translunar coast period 3	27.947	33.947	6.000	41 Lunar orbit coast	222.570	245.900	23.330
12 Navigation sighting	33.947	34.947	1.000	42 Lunar orbit navigation period 3	245.900	248.900	3.000
13 Translunar coast period 4	34.947	46.947	12.000	43 Transearth injection	248.900	245.940	0.040
14 Navigation sighting	46.947	49.947	3.000	44 Transearth midcourse correction 1	248.940	251.940	3.000
15 Translunar coast period 5	49.947	63.780	13.833	45 Transearth coast period 1	251.940	271.940	20.000
16 Third midcourse correction	63.780	65.780	2.000	46 Transearth midcourse correction 2	271.940	273.940	2.000
17 Translunar coast period 6	65.780	71.780	6.000	47 Transearth coast period 3	273.940	281.360	7.420
18 Navigational sighting	71.780	72.780	1.000	48 Navigational sightings	281.360	282.360	1.000
19 Translunar coast period 7	72.780	78.780	6.000	49 Transearth coast period 4	282.360	297.200	14.840
20 Fourth midcourse correction	78.780	81.780	3.000	50 Navigational sightings	297.200	300.200	3.000
21 Lunar orbit injection at 80 nautical miles	81.780	81.870	0.090	51 Transearth coast period 5	300.200	315.100	14.900
22 Lunar orbit navigation period 1	81.870	83.870	2.000	52 Transearth midcourse correction 3	315.100	317.100	2.000
23 Lunar orbit coast	83.870	122.870	39.000	53 Transearth coast period 6	317.100	324.520	7.420
24 Crew transfer and check out	122.870	123.870	1.000	54 Navigational sighting	324.520	325.520	1.000
25 Injection into "Equal Period Orbit"	123.870	123.880	0.010	55 Transearth coast period 7	325.520	332.940	7.420
26 Transfer to 50, 000 feet	123.880	124.370	0.490	56 Transearth midcourse correction 4	332.940	335.930	2.990
27 Equal period orbit stay	124.370	130.020	5.650	57 Service module separation	335.930	335.940	0.010
28 Perilune retro (50, 000 to 1, 000 feet)	130.020	130.120	0.100	58 Entry	335.940	336.273	0.333
29 Hover, translation, and touchdown	130.120	130.160	0.040	59 Deploy drogue chute through landing	336.273	336.410	0.137
30 LEM lunar surface stay period 1	130.160	145.660	15.500	60 Postlanding		408.410	72.000

[redacted]

RESULTS OF MISSION SIMULATION

Three Monte Carlo mission simulations were completed for the purpose of evaluating total system mission success reliability and to establish a new set of subsystem reliability requirements based on the lunar orbital rendezvous mission. The results of these studies are presented in Tables 3, 4, and 5. Table 3 contains an estimate of Apollo system, subsystem, and component reliabilities based on present state-of-the-art failure rates. Table 4 presents the results of a similar analysis based on high-reliability parts in the mechanical subsystem and high-reliability (comparable to Minuteman) parts in all the electronic systems, except for the guidance and navigation system, for which Massachusetts Institute of Technology-furnished Polaris parts failure rates were used. Table 5 contains the results of the analysis based on high-reliability parts for all subsystems, including the guidance and navigation. The results of Tables 4 and 5 comprise the new apportionments for the spacecraft based on the lunar orbital rendezvous mission, since these results indicate the required spacecraft reliability of 0.965. The results in Tables 4 and 5 include sufficient spares to bring the estimated reliability up to the required level, while those of Table 3 include only those spares which are presently planned for inclusion within the spacecraft.

The input data for the run the results of which are presented in Table 3 utilized failure rates for the present state-of-the-art. These failure rates are consistent with those being achieved on military aircraft for the mechanical and electrical subsystems and are based on MIL standard parts for electronic subsystems, except for the guidance and navigation subsystem, which utilizes Massachusetts Institute of Technology data. For the booster and the lunar excursion module, the work statement reliability requirements of 0.95 and 0.984, respectively, were used. The spares utilized were those which are presently planned for the spacecraft and have a total weight of 101.7 pounds.

There were 5301 missions simulated for this run, including the C-5 booster and the spacecraft, resulting in a total reliability of 0.69. The spacecraft alone is estimated to be approximately 0.74. The confidence interval was ±.005 at the 90 percent confidence level. Individual component and system failures are tabulated in column 3 of Table 3. Column 4 tabulates the reliability of the subsystems and components, computed from the number of failures.

The input data for the run presented in Table 4 were adjusted to result in a total spacecraft reliability of 90 percent, including the boosters and

Table 3. Apollo Lunar Orbital Rendezvous Mission, Predicted State-of-the-Art Reliability

Subsystem	Component	No. of Failures Causing Mission Abort	Estimated Reliability	Predicted State of the Art
Structural and mechanical	Command module structure			0.99989
	Service module structure			0.999947
	Service module - command module separation			0.999998
	Lunar excursion module adapter separation			0.999995
	Lunar excursion module structure			0.999994
				0.999956
Launch escape system	Launch escape tower			0.99
	Launch escape system separation			
Electric power system		38	0.989793	0.9941
	Battery and inverters	15	0.995945	
	Fuel cell and busses	23	0.993797	
Earth land				0.99994
Cryogenic, storage				0.9965
	Oxygen			0.9980
	Hydrogen			0.9985
Service module Reaction control system	Service module - reaction control system translunar	31	0.991657	0.999357
		26	0.992993	
	Service module - reaction control system transearth	5	0.9986	
Command module reaction control system		28	0.992458	0.993473
	Command module - reaction control system - total mission	8	0.997833	
	Command module - reaction control system - entry	20	0.994601	
Lunar excursion module*		60	0.984	0.984
Environmental control subsystem			0.979792	0.9805
	O_2 supply	2	0.999457	0.99957
	H_2O glycol circuit	31	0.991657	0.99190
	Water supply	4	0.998915	
	Command module pressure and temperature	3	0.999186	0.99932
	Recirculation blower	3	0.999186	
	Pressure suit circuit	33	0.991124	0.9910
Service propulsion system	Fuel system engines	135	0.964659	0.96470
Boosters*		194	0.950000	0.950000
Electronics			0.847712	0.673030
Stabilization and control			0.87996	0.85029
Guidance and navigation			0.97049	0.95313
Communications and data			0.99349	0.84151
Instrumentation			0.99844	0.78687

* Apportioned reliabilities

Table 4. Apollo Lunar Orbital Rendezvous Mission, Apportioned Reliability (Polaris Components for Guidance and Navigation)

Subsystem	No. of Failures Causing Mission Abort	Estimated Reliability	Apportioned Lunar Orbital Rendezvous	Apportioned Direct Flight
Structural and mechanical	2	0.999926	0.999926	0.997
Launch escape system	7	0.999741	0.999989	0.999989
Electric power system	31	0.998854	0.9989	0.9955
Earth land			0.99994	0.99994
Cryogenic storage			0.9989	0.9989
Service module reaction control system	16	0.999409	0.999409	0.996
Command module reaction control system			0.999969	0.99996
Lunar excursion module	461	0.983235	0.984	
Environmental control subsystem	63	0.997675	0.997675	0.9935
Service propulsion system	34	0.998744	0.99977	0.99977
Boosters	1477	0.948200	0.95	0.95
Electronics			0.988901	0.9771947
Stabilization and control		0.98972	0.987735	0.995
Guidance and navigation		0.98864	0.994558	0.994
Communications and data		0.99496	0.998502	0.9925*
Instrumentation		0.99857	0.9700011*	0.9955

* Not Included for Mission Success

Table 5. Apollo Lunar Orbital Rendezvous Mission, Apportioned Reliability (Minuteman Components for Guidance and Navigation)

Subsystem	No. of Failures Causing Mission Abort	Estimated Reliability	Apportioned Lunar Orbital Rendezvous	Apportioned Direct Flight
Structural and mechanical			0.999926	0.997
Launch escape system	6	0.99977	0.999989	
Electric power system	36	0.998665	0.9989	0.9955
Earth land	1	0.999962	0.99994	0.99994
Cryogenic storage			0.9989	0.9989
Service module reaction control system	18	0.999332	0.999409	0.996
Command module reaction control system			0.999969	0.99996
Lunar excursion module	424	0.984510	0.984	
Environmental control subsystem	56	0.997926	0.997675	0.9935
Service propulsion system	39	0.998554	0.99977	0.999977
Boosters	1496	0.947407	0.95	0.95
Stabilization and control		0.975	0.978641	
Guidance and navigation		0.996	0.996381	
Communications and data		0.995	0.995831	
Instrumentation		0.999	0.998994	

lunar excursion module. This adjustment was made by considering the levels of reliability practical for the mechanical parts. For the electronic parts, Minuteman failure rates were used for the stabilization and control, communications and data, and the instrumentation subsystems, since this type of part is presently available for utilization in the Apollo spacecraft. The failure rates used for the guidance and navigation subsystem were those received from Massachusetts Institute of Technology. They are consistent with the type of parts presently planned for utilization. For the boosters and the lunar excursion module the Work Statement reliability requirements of 0.95 and 0.984, respectively, were utilized. Based on the results of previous runs, the number of spares necessary to bring the reliabilities up to the required level was estimated. Then the number of spares was included in the Monte Carlo program. This run verifies that these subsystem reliabilities and spares estimations are consistent with the overall spacecraft requirement.

There were 27,702 missions simulated for this run, resulting in a total reliability of 0.908. The confidence interval was ±.0028 at the 90 percent confidence level. Individual component and subsystem failures are tabulated in Column 3 of Table 4. Column 4 tabulates the reliability of the subsystems computed from this number of failures. Column 5 presents the apportioned values based on the results of the run and the analyses presented in the subsystem section. For comparison, the previous subsystem requirements are listed in Column 6. The spares required to achieve these levels are discussed in the individual subsystem sections. The total weight for these spares is 139.1.

A similar procedure was used for the input data of the run represented in Table 5, except that high-reliability parts failure rates were used in the guidance and navigation subsystem. The purpose of this run was to determine the difference in the weight of spares required, when the lower-reliability parts are utilized in the guidance and navigation subsystem. For this run, 30,000 missions were simulated, resulting in a total spacecraft reliability of 0.898 percent. The confidence interval for the result was ±.0028 at the 90 percent confidence level. The failures that occurred, reliability for the various subsystems, and the previously apportioned reliabilities are listed in the table. The total weight of the spares utilizing this set of failure rates is 114.1.

The total effect of the reapportionment in this run was to lower the reliability requirements on the stabilization and control and guidance subsystems and raise the requirements on the instrumentation, communications and data, and mechanical subsystems. The requirements were raised on the mechanical subsystems, since the latest analyses indicate that a higher level of reliability can be achieved than was initially indicated. The communication and data and instrumentation subsystem requirements were raised,

because equipment has been removed since the initial apportionment, thus making it possible to meet a higher reliability level. The stabilization and control subsystem requirement was lowered from 0.995 to 0.990, which is equivalent to five failures per 1000 missions versus ten failures per 1000 missions. A similar reduction was possible in the guidance and navigation subsystem, lowering the requirement from 0.994 to 0.989, which is equivalent to six failures per 1000 missions versus eleven failures per 1000 missions. This reapportionment and adjustment in the number of spares has increased the total spares weight from 102 pounds to 139 pounds.

If Minuteman parts are used in the guidance and navigation subsystem, (Table 5), a modification in the stabilization and control and guidance and navigation subsystems requirements could be made. Utilizing these parts, the guidance and navigation requirement would be raised to 0.996, or four failures per 1000 missions, and the stabilization and control subsystem requirement would be reduced to 0.975, or 25 failures per 1000 missions. One of the major reasons that the guidance and navigation requirement would be higher than the stabilization and control subsystem is that the stabilization and control subsystem operates for only a portion of the mission. These requirements reflect an optimum configuration of spares; i.e., the spares are inserted where the greatest reliability increase per pound results. This configuration would allow a decrease in the total weight of spares from 139 pounds to 114 pounds.

A detailed discussion of the subsystems is contained in the subsystem analysis section.

PHASE RELIABILITY

Table 6 presents the reliabilities for the various phases of the mission using high-reliability mechanical parts, Polaris parts for the guidance and navigation subsystem, and Minuteman parts on the other electronic subsystems. Mission phases 1, 2, 3, and 5 have low reliability, primarily due to the boosters which have a combined reliability of 0.95. Mission phase 4, the earth orbit stay period, has a relatively low reliability, since all spacecraft equipment is checked out during this period. Further analysis of the mission phases shows low reliability during those periods of navigational sightings and midcourse corrections in which the electronic and propulsion equipment is under high stress. Expected low reliabilities occur during the lunar operational phases, due to the requirement of operating both spacecraft and lunar excursion module equipment. The high reliability in the transearth phases of the mission is the result of utilizing all the back-up modes available in the spacecraft, the philosophy being that a safe return after the successful completion of lunar operations is a mission success.

Table 6. Reliability by Phase

No.	Phase	Failures	Reliability	No.	Phase	Failures	Reliability
1	Launch to 100,000 feet	315	0.989465	31	Command module navigational sightings	32	0.998836
2	100,000 feet to jettison of launch escape tower	162	0.994524	32	Lunar excursion module lunar surface stay (Period 2)	57	0.997925
3	Launch escape tower jettison to earth orbit injection	818	0.972200	33	Lunar ascent to circular orbit at 50,000 ft	33	0.998796
4	Earth orbit stay (3 orbits)	229	0.991994	34	Ascent to 80 nm	16	0.999415
5	Translunar injection	203	0.992846	35	Rendezvous and docking	156	0.994299
6	Translunar coast cisearth (Period 1)	0	1.0000	36	Lunar orbit navigations	2	0.999927
7	Lunar excursion module transfer	5	0.999822	37	Lunar orbit coast	5	0.999816
8	1st midcourse correction	55	0.998047	38	Lunar orbit navigation sightings	1	0.999963
9	Translunar coast (Period 2)	16	0.999430	39	Lunar orbit coast	10	0.999632
10	2nd midcourse correction	47	0.998327	40	Lunar orbit navigation sightings	2	0.999926
11	Translunar coast (Period 3)	6	0.999786	41	Lunar orbit coast	7	0.999742
12	Navigation sighting	17	0.999393	42	Lunar orbit navigation (Period 3)	3	0.999889
13	Translunar coast (Period 4)	16	0.999429	43	Transearth injection	0	-
14	Navigation sighting	73	0.997394	44	Transearth midcourse correction 1	11	0.999595
15	Translunar coast (Period 5)	25	0.999105	45	Transearth coast (Period 1)	1	0.999963
16	3rd midcourse correction	53	0.998101	46	Transearth midcourse correction 2	8	0.999705
17	Translunar coast (Period 6)	13	0.999533	47	Transearth coast (Period 3)	1	0.999963
18	Navigational sighting	21	0.999245	48	Navigational sightings	4	0.999852
19	Translunar coast (Period 7)	9	0.999676	49	Transearth coast (Period 4)	5	0.999815
20	4th midcourse correction	78	0.997196	50	Navigational sightings	10	0.999631
21	Lunar orbit injection at 80 nm	0	1.0000	51	Transearth coast (Period 5)	4	0.999852
22	Lunar orbit navigation (Period 1)	48	0.998269	52	Transearth midcourse correction 3	7	0.999742
23	Lunar orbit coast	46	0.998338	53	Transearth coast (Period 6)	4	0.999852
24	Crew transfer and checkout	8	0.999710	54	Navigational sighting	3	0.999889
25	Injection into equal period orbit	0	1.0000	55	Transearth coast (Period 7)	1	0.999963
26	Transfer to 50,000 ft	8	0.999710	56	Transearth midcourse correction 4	8	0.999705
27	Equal period orbit stay	54	0.998045	57	Space module separation	0	1.0000
28	Perilune retro (50,000 ft to 1000 ft)	2	0.999927	58	Entry	61	0.997750
29	Hover, translation and touchdown	2	0.999927	59	Deploy drogue chute through landing	17	0.999371
30	Lunar excursion module lunar surface stay (Period 1)	67	0.997570				

High-reliability mechanical parts
Polaris parts for guidance and navigation subsystems
Minuteman parts for other electronic subsystems

The high number of failures occurring in the entry portion of the mission are primarily in the electronic equipment, indicating that improvement is necessary in this area. Even though the mission requirement for the spacecraft of 0.90 reliability is achieved, failures in this phase of the mission result in crew loss and subsequently effect the crew safety reliability requirement for the spacecraft of 0.999.

The reliability degradation curve (Figure 4) shows the probability of reaching certain phases of the mission. In addition, the degree of discontinuity of the curve indicates the criticality of the mission phases. This curve further shows the undesirability of performing a serial reliability analysis of the Apollo spacecraft, since the degree of discontinuity at various phases of the mission precludes the application of an exponential failure density distribution to the Apollo spacecraft in the Apollo mission.

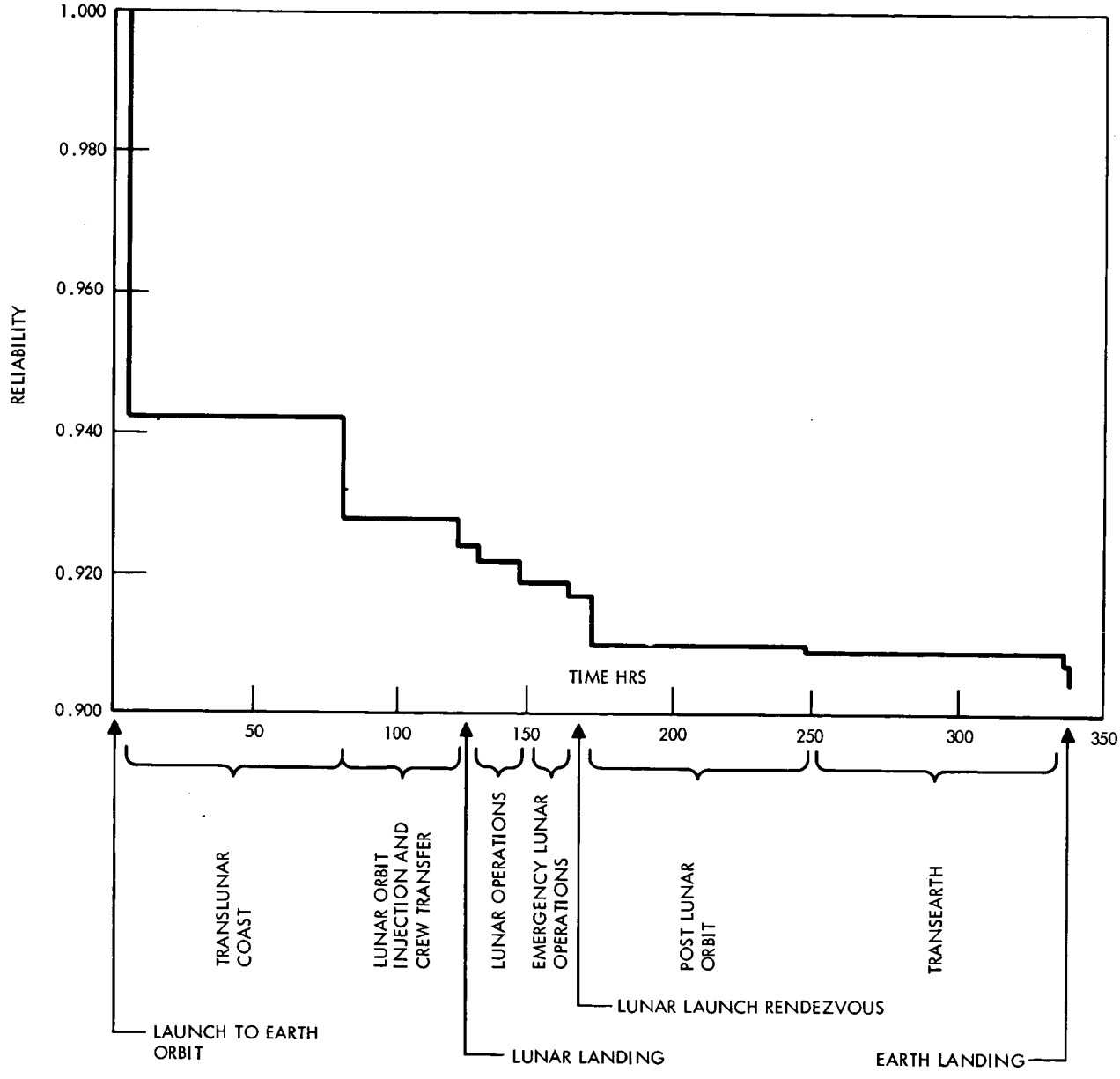


Figure 4. Reliability Degradation Versus Phase Time

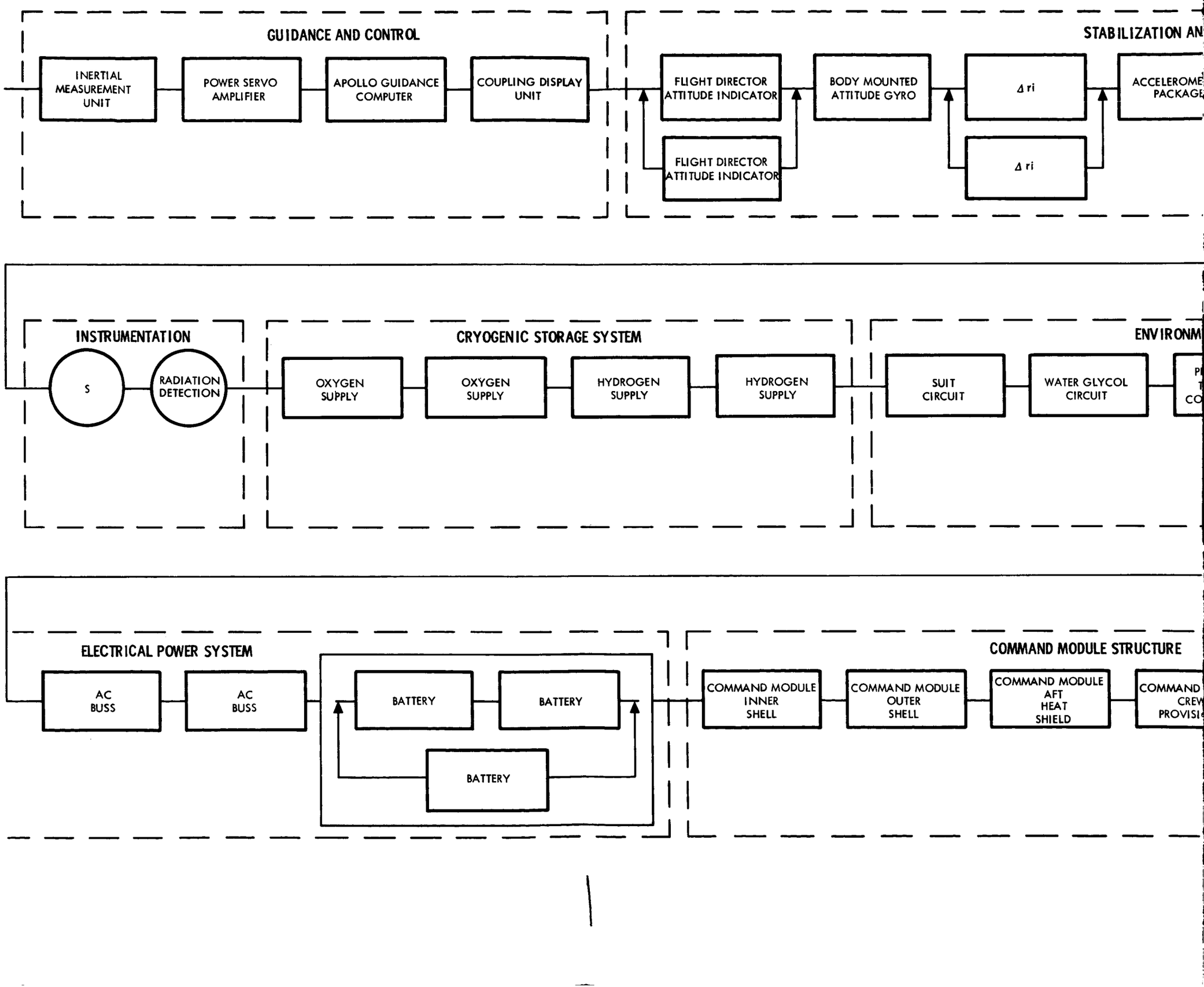
SUMMARY

This section contains the logic diagrams that define the equipment necessary for mission success. The diagrams are presented for the various phases of the mission, as described in the previous section, Mission Phase Time Line. The diagrams show the equipments that must operate successfully to preclude an abort up to lunar excursion module rendezvous and docking. From this time to earth landing, any combinations of equipments which will allow successful return to earth are considered sufficient for mission success and are so shown on the diagrams. Blocks that appear in a continuous chain are serial equipments; that is, each must operate properly in order to successfully complete the mission. Equipments shown in parallel are redundant to one another; i.e., either path may be followed for successful mission completion. Where parallel equipments are connected with arrowheads, the alternate component is either an on-board spare or is in stand-by redundancy, which means it does not operate until the primary component fails. While the total analysis contains approximately 100 diagrams, only typical examples are presented in this report.

LOGIC DIAGRAMS AND SUBSYSTEM OPERATION

The first block diagram shown, Figure 5, covers the time phase of earth orbit. During this phase, it is required that all equipment operate without failure or that a spare part be available to replace a failed component; hence, all of the navigation and guidance, stabilization and control, and telecommunications equipment that operates during this phase is shown in series. Although this equipment is not being utilized to perform the guidance and control function, which is performed by S-IV equipment, these parts are operating, and their failure would cause a mission abort. Spare parts carried on board would replace the flight director attitude indicator, the delta velocity indicator, the audio center or the VHF/AM transceiver, and one of the three environmental control system units. Hence, in the event of failure of only these modules, the module would be replaced and the mission continued. The service module - service propulsion system, service module - reaction control system, command module - reaction control system, earth landing system, and command module - forward heat shield are not shown in Phase I because they are in a quiescent state. Redundancies are available for crew safety in the environmental control system, electrical power system, and cryogenic storage system; however, these redundancies cannot be considered for mission success. Should one of the redundant crew safety components fail in Phase I, the mission would be aborted. The service module to command module separation mechanism is not called upon to operate within Phase I and is left out of the structure.

Figure 6 indicates the equipment necessary during the translunar phase. The most necessary function performed by the electronic equipment during this phase is that of midcourse correction. The first function is that of inertial measurement units alinement. This is done by placing the spacecraft in the automatic control mode, rotating the spacecraft to a position from which a stellar sighting can be made, stabilizing the spacecraft in this position, and sighting a star with the sextant. The spacecraft is then held in a stabilized position while the inertial measurement unit is being automatically erected, utilizing the guidance and navigation equipment. Hence, the navigational and control equipments, except the spare parts for the electronic control assemblies and the three-axis rotational control, are all shown in series for this function, since an abort would be indicated in the event of a failure. The DSIF transponder and audio center are also shown in series because they provide backup for the navigational equipment. In the event of their failure, the mission would be aborted, since no additional backup equipment would be available. Spares are carried for the DSIF equipment and the audio center. Further redundancy exists within the


GUIDANCE AND CONTROL
INERTIAL MEASUREMENT UNIT
POWER SERVO AMPLIFIER
APOLLO GUIDANCE COMPUTER
COUPLING DISPLAY UNIT
STABILIZATION AN
FLIGHT DIRECTOR ATTITUDE INDICATOR
BODY MOUNTED ATTITUDE GYRO
Δ ri
ACCELEROME PACKAGE
FLIGHT DIRECTOR ATTITUDE INDICATOR
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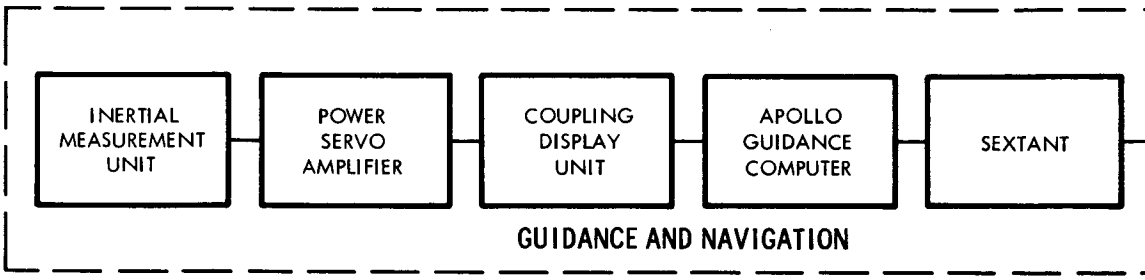
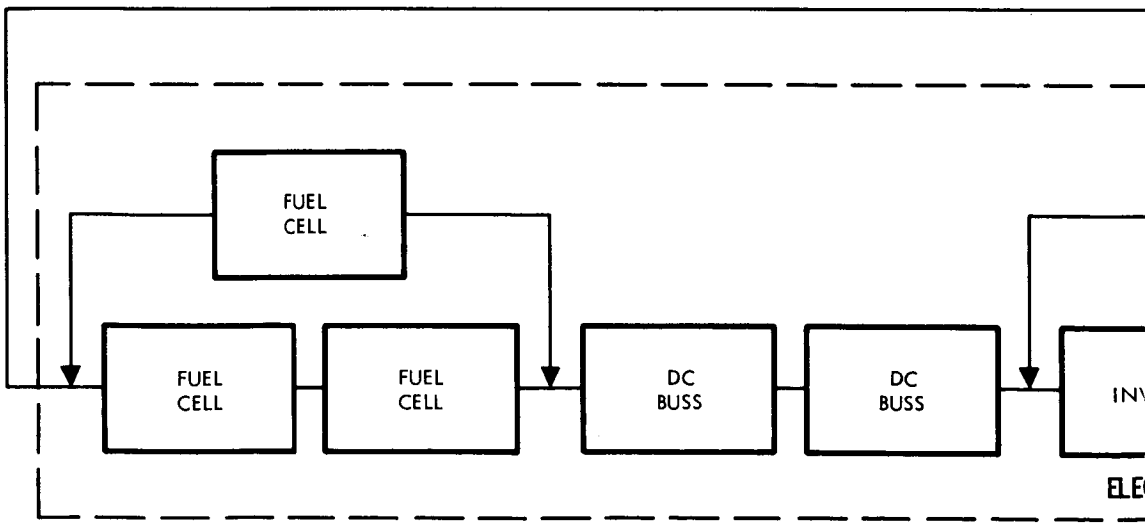
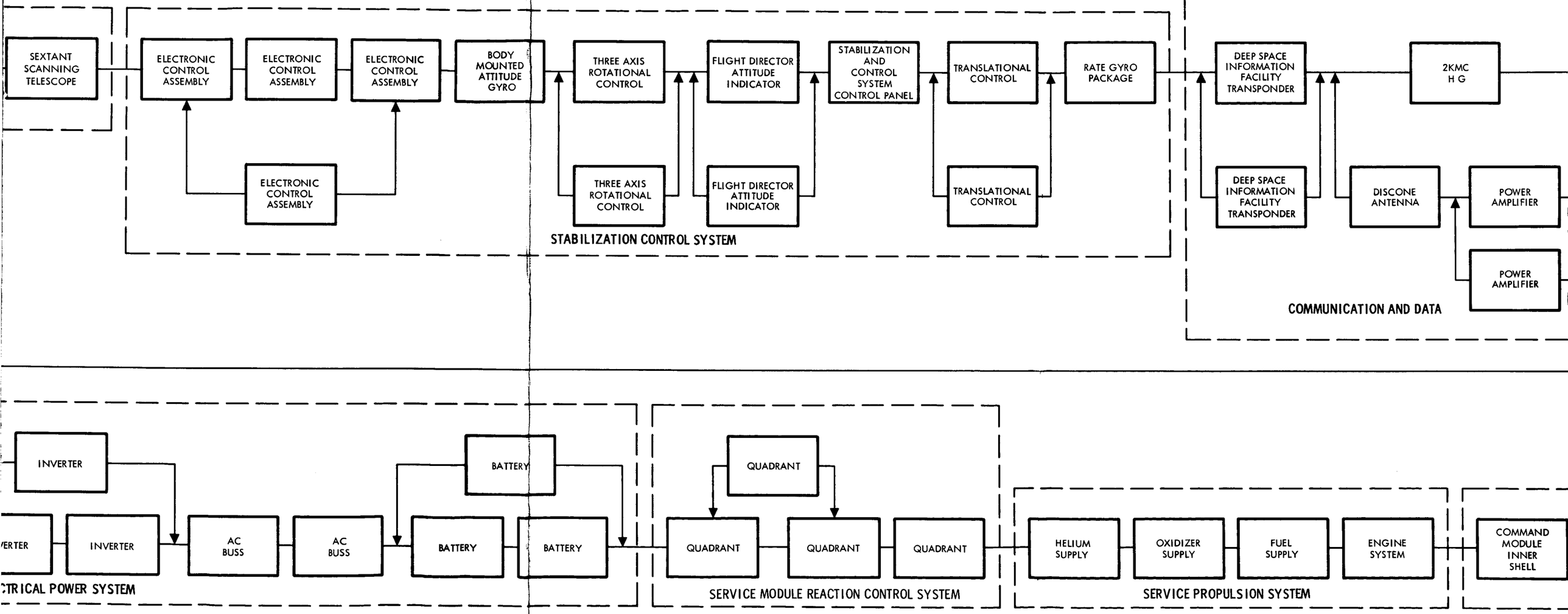
INSTRUMENTATION
S
RADIATION DETECTION
CRYOGENIC STORAGE SYSTEM
OXYGEN SUPPLY
OXYGEN SUPPLY
HYDROGEN SUPPLY
HYDROGEN SUPPLY
ENVIRONM
SUIT CIRCUIT
WATER GLYCOL CIRCUIT
ELECTRICAL POWER SYSTEM
AC BUSS
AC BUSS
BATTERY
BATTERY
BATTERY
COMMAND MODULE STRUCTURE
COMMAND MODULE INNER SHELL
COMMAND MODULE OUTER SHELL
COMMAND MODULE AFT HEAT SHIELD
COMMAND CREW PROVISI

Figure 5. Phase I - Launch Through Earth Orbit


INERTIAL MEASUREMENT UNIT
POWER SERVO AMPLIFIER
COUPLING DISPLAY UNIT
APOLLO GUIDANCE COMPUTER
SEXTANT
GUIDANCE AND NAVIGATION


FUEL CELL
FUEL CELL
FUEL CELL
DC BUSS
DC BUSS
INV
ELE


SEXTANT SCANNING TELESCOPE
ELECTRONIC CONTROL ASSEMBLY
ELECTRONIC CONTROL ASSEMBLY
ELECTRONIC CONTROL ASSEMBLY
ELECTRONIC CONTROL ASSEMBLY
BODY MOUNTED ATTITUDE GYRO
THREE AXIS ROTATIONAL CONTROL
THREE AXIS ROTATIONAL CONTROL
FLIGHT DIRECTOR ATTITUDE INDICATOR
FLIGHT DIRECTOR ATTITUDE INDICATOR
STABILIZATION AND CONTROL SYSTEM CONTROL PANEL
TRANSLATIONAL CONTROL
TRANSLATIONAL CONTROL
RATE GYRO PACKAGE
STABILIZATION CONTROL SYSTEM
DEEP SPACE INFORMATION FACILITY TRANSPONDER
DEEP SPACE INFORMATION FACILITY TRANSPONDER
2KMC H G
DISCONE ANTENNA
POWER AMPLIFIER
POWER AMPLIFIER
COMMUNICATION AND DATA
INVERTER
INVERTER
INVERTER
AC BUSS
AC BUSS
BATTERY
BATTERY
BATTERY
CTRICAL POWER SYSTEM
QUADRANT
QUADRANT
QUADRANT
QUADRANT
SERVICE MODULE REACTION CONTROL SYSTEM
HELIUM SUPPLY
OXIDIZER SUPPLY
FUEL SUPPLY
ENGINE SYSTEM
SERVICE PROPULSION SYSTEM
COMMAND MODULE INNER SHELL

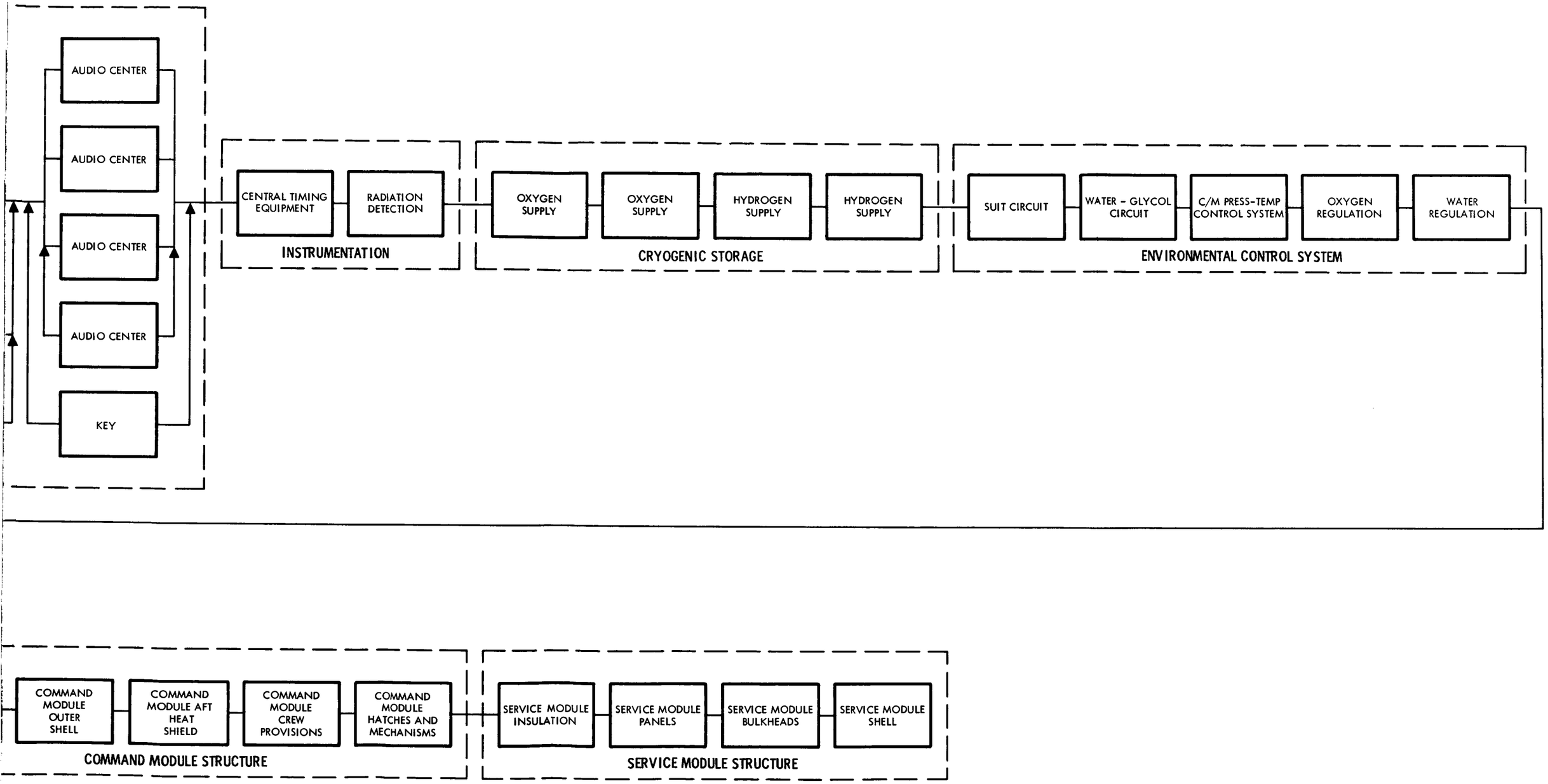
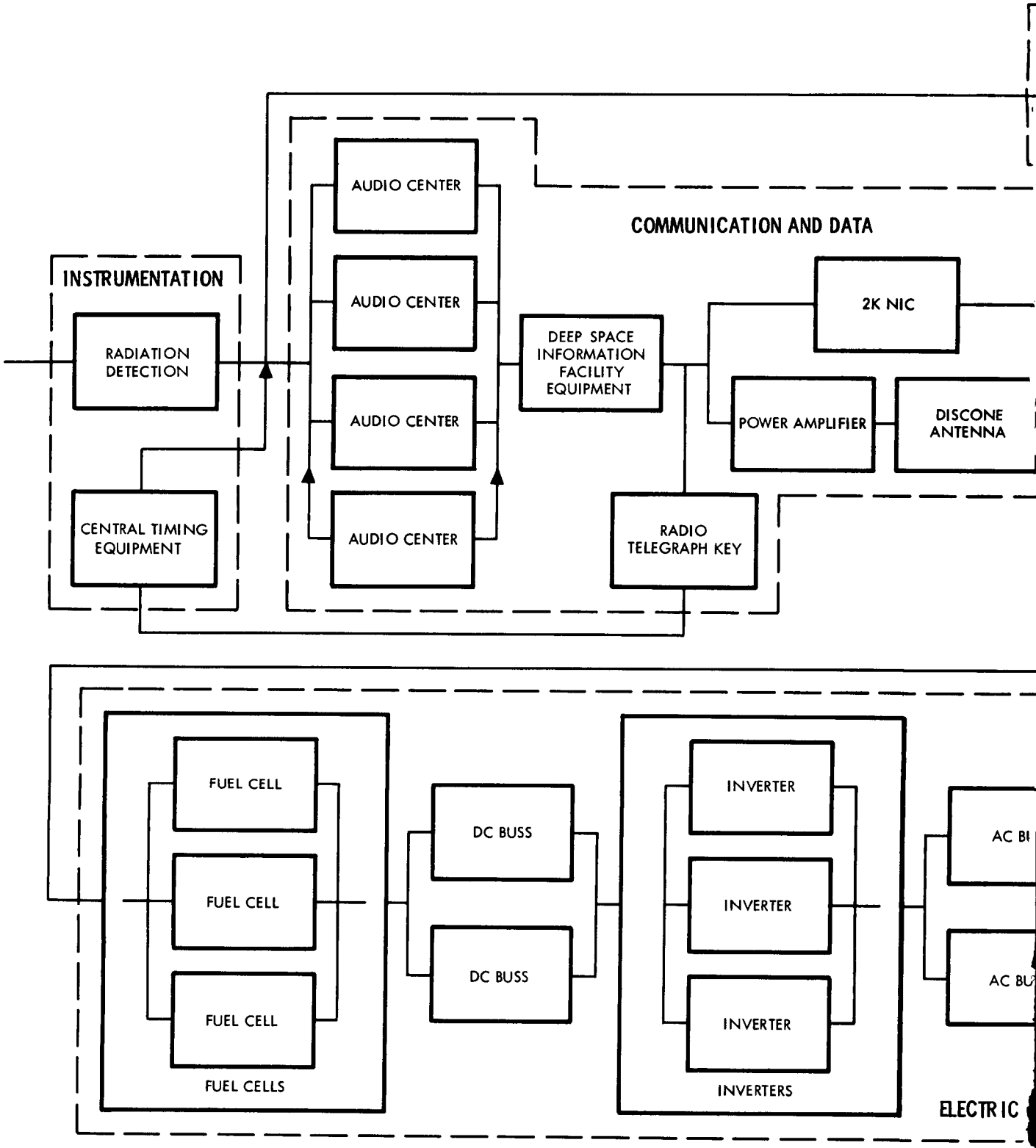


Figure 6. Phase II - Translunar Injection/LEM Recovery

audio center, a unit being provided for each crew member, any one of which can be utilized for communications. The 2kmc high-gain antenna is the normal mode of operation, but this equipment is backed up by the power amplifier and the discone antenna. A spare power amplifier is also provided. The service module - service propulsion system is fired for midcourse correction and lunar orbit injection. Subsequent time within Phase II is considered crew safety for this system (Phase III diagram). All other systems, except service module - reaction control system, must operate until lunar excursion module ascent from the lunar surface to qualify for mission success. The command module - reaction control system, earth landing system, service module - command module separation mechanism, and command module - forward heat shield are still omitted. Even though environmental control system redundancies cannot be used during this phase, the load is reduced to one-third after lunar excursion module separation from command module. The service module - reaction control system has its most difficult task during lunar excursion module rendezvous and docking. Not only must attitude control be maintained, but also translation in all directions must be available. More complete definition of the docking method is required before relaxation of these functions can be allowed. Mission success logic must then apply to the service module reaction control system through lunar excursion module docking. Helium regulators within the service module - service propulsion system are not considered an abort item, because no means is available for detecting a failed closed condition.

The third phase (Figure 7), covers a time period from lunar excursion module rendezvous to entry. Since no aborts are possible during this phase, all modes of operation which are possible for navigation and guidance are considered adequate for mission success. This block diagram does not show the primary modes of operation, which are identical to those utilized in the translunar phase, but only shows the minimum equipments necessary to successfully complete the mission.

The first function performed during this phase is the establishment of an angular inertial reference, using either the inertial measurement unit or the body-mounted attitude gyro. When the inertial measurement unit is being utilized, the equipment shown in series with it is also required. This includes the coupling display unit, a power and servo amplifier, and the Apollo guidance computer. If the body-mounted attitude gyro is being used because of an inertial measurement unit failure, it is also necessary to utilize the flight director attitude indicator. During this alinement process, the automatic stabilization system will normally be used, which includes the electronic control assembly, stabilization and control system control panel, three axis rotational control, and the rate gyro package. The three axis rotational control is provided with a spare, as is the electronic control assembly. In the event of a failure of the electronic control assembly and its spare, the manual emergency control is used.


AUDIO CENTER
COMMUNICATION AND DATA
INSTRUMENTATION
AUDIO CENTER
2K NIC
RADIATION DETECTION
DEEP SPACE INFORMATION FACILITY EQUIPMENT
AUDIO CENTER
POWER AMPLIFIER
DISCONE ANTENNA
CENTRAL TIMING EQUIPMENT
AUDIO CENTER
RADIO TELEGRAPH KEY
FUEL CELL
DC BUSS
INVERTER
AC BI
FUEL CELL
INVERTER
DC BUSS
FUEL CELL
INVERTER
AC BU
FUEL CELLS
INVERTERS
ELECTRIC

GUIDANCE AND NAVIGATION

INERTIAL MEASUREMENT UNIT

COUPLING DISPLAY UNIT

POWER SERVO AMPLIFIER

APOLLO GUIDANCE COMPUTER

SCANNING TELESCOPE

SCANNING TELESCOPE

GROUND OPERATIONAL SUPPORT SYSTEM

FLIGHT DIRECTOR ATTITUDE INDICATOR

FLIGHT DIRECTOR ATTITUDE INDICATOR

DIFFERENTIAL VELOCITY INDICATOR

DIFFERENTIAL VELOCITY INDICATOR

BODY MOUNTED ATTITUDE GYRO

ACCELEROMETER PACKAGE

STABILIZATION AND CONTROL

ELECTRONIC CONTROL ASSEMBLY

ELECTRONIC CONTROL ASSEMBLY

ELECTRONIC CONTROL ASSEMBLY

ELECTRONIC CONTROL ASSEMBLY

MANUAL EMERGENCY CONTROLS

THREE AXIS ROTATIONAL CONTROL

THREE AXIS ROTATIONAL CONTROL

TRANSLATIONAL CONTROL

TRANSLATIONAL CONTROL

STABILIZATION AND CONTROL SYSTEM CONTROL PANEL

RATE GYRO PACKAGE

OXYGEN SUPPLY

OXYGEN SUPPLY

CRYOGENIC STORAG

BATTERY

BATTERY

BATTERY

BATTERIES

POWER SYSTEM

QUADRANT

QUADRANT

QUADRANT

QUADRANT

SERVICE MODULE REACTION CONTROL SYSTEM

HELIUM SUPPLY

OXIDIZER SUPPLY

FUEL SUPPLY

SERVICE MODULE PROPULSION SYSTEM

VALVE SET OXYGEN AND FUEL

VALVE SET OXYGEN AND FUEL

VALVE SET OXYGEN AND FUEL

VALVE SET OXYGEN AND FUEL

PITCH GIMBAL MOTOR, CLUTCH AND GEAR SET

PITCH GIMBAL MOTOR, CLUTCH AND GEAR SET

YAW GIMBAL MOTOR, CLUTCH AND GEAR SET

YAW GIMBAL MOTOR, CLUTCH AND GEAR SET

ENGINE SYSTEM

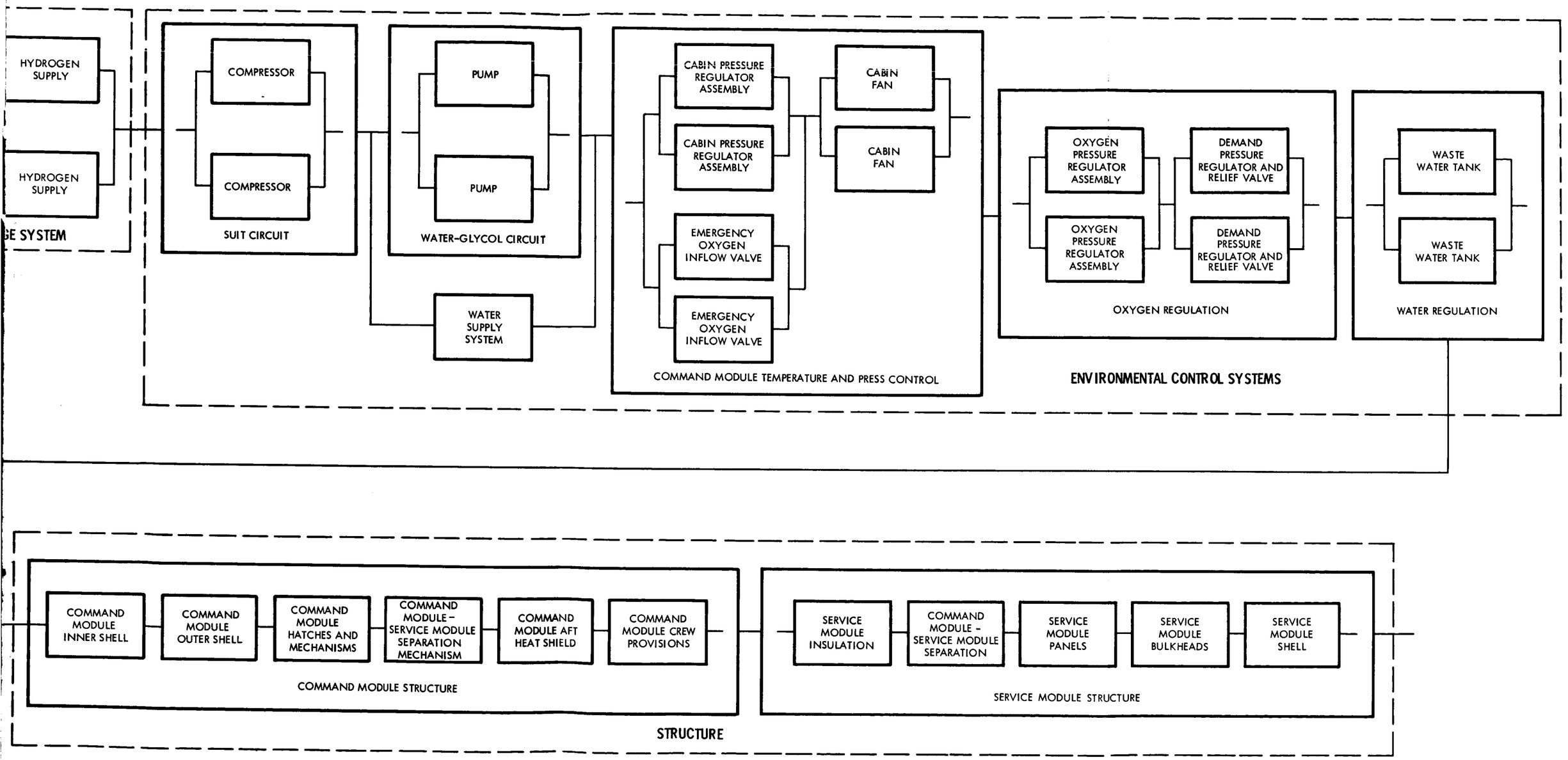


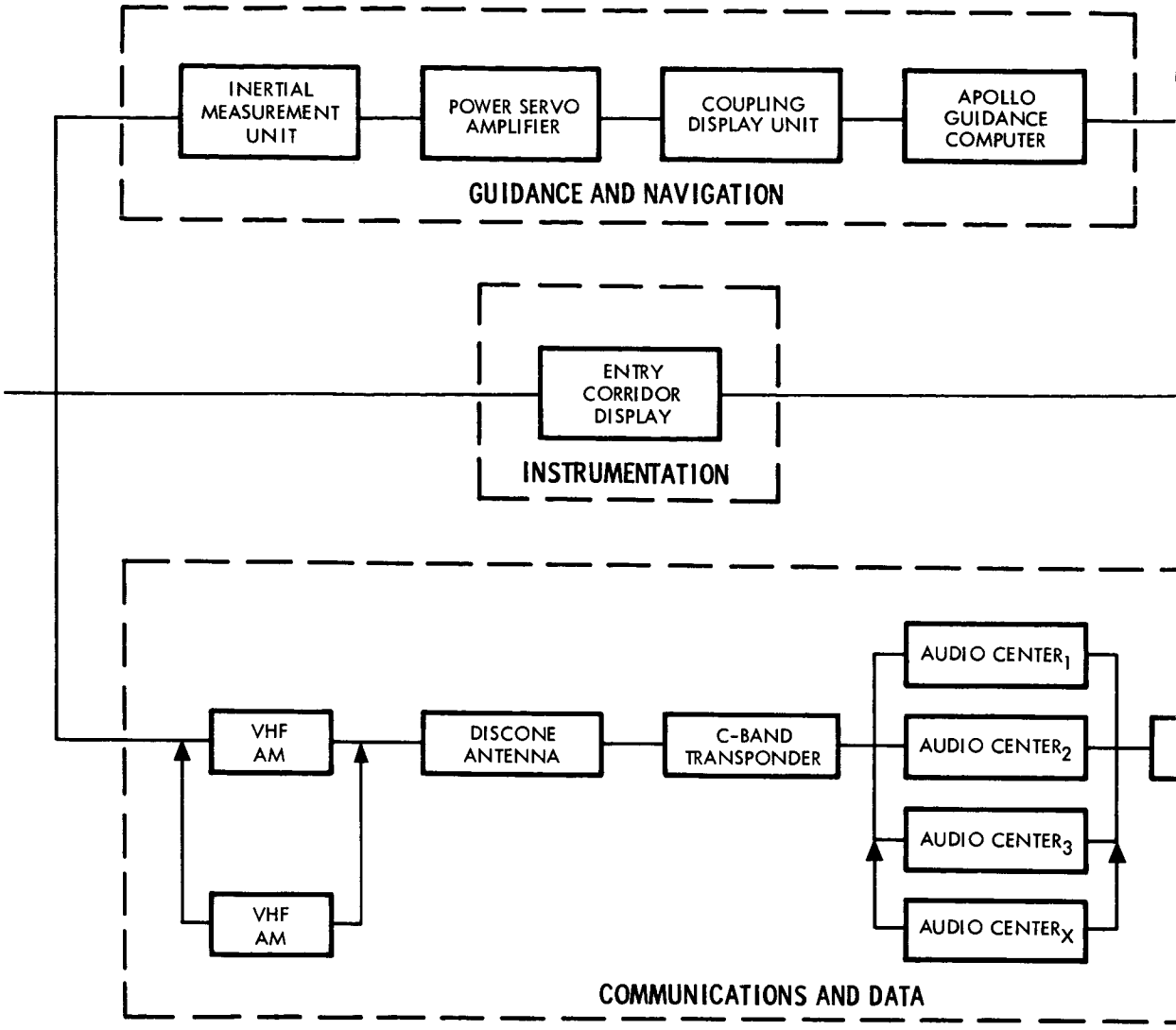
Figure 7. Phase III - Transearth Through Separation

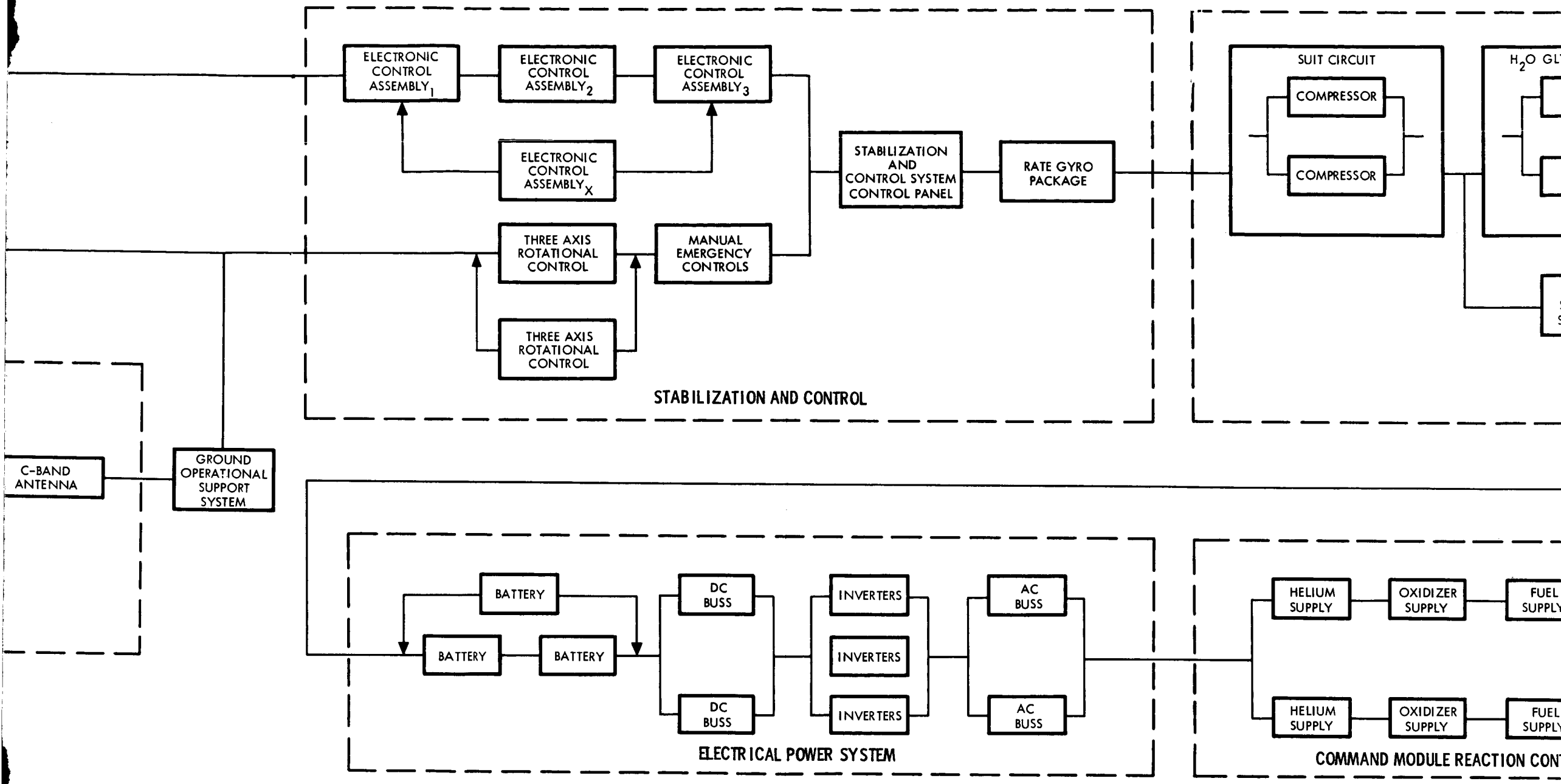
The next function to be performed is the determination of the required ΔV for midcourse correction. This could be determined either through the use of the guidance and navigation equipment, including the inertial measurement unit, Apollo guidance computer, coupling display unit, and power servo amplifier, or communication with earth through the DSIF and GOSS equipment. Communication with earth can be achieved through the discone antenna, utilizing the power amplifier. Also required for communication is one of the three audio centers, or if these are failed, the key used in conjunction with the central timing equipment is necessary, since timing data cannot be transmitted via cw.

The next function, the rotation of the spacecraft prior to application delta velocity, is achieved, using either the automatic or manual stabilization and control equipment described for the inertial measurement unit alinement phase. These equipments are also utilized for stabilization of the spacecraft during the ΔV application. The control of the ΔV application is normally through the use of the inertial measurement unit, Apollo guidance computer, and associated equipment. In the event of failure of this equipment, the ΔV's can be manually applied, using the stabilization and control system accelerometer package and the ΔV indicator. Two propulsion means are available for this ΔV application: (1) fine control with aft firing service module - reaction control engines and (2) course control with the service module - service propulsion system. The magnitude of correction by the service module - reaction control system is limited to the availability of the propellants in tanks and quadrants still working. The service module - service propulsion system is the only means for transearth injection and is dependent, in general, on the operation of single elements; i.e. chamber, injector, flexible piping, fuel tank, oxidizer tank, and helium tanks. Many items can now be considered in full redundancy for crew safety, because any means of safe return is acceptable. For example, one main propellant valve set (oxidizer and fuel) can fail, either open or closed, and another valve set will be available for functional replacement. One quadrant of the service module - reaction control system can fail and the attitude control, propellant settling, and separation functions will be retained. Translation in one direction will be lost, but the need for three-axis translation will be past. Opposite quadrants must be retained for settling propellants or separating command module from service module, and one adjacent quadrant must be available for pitch or yaw. The electrical power system becomes entirely redundant; any one fuel cell out of three, either d-c bus; any one inverter out of three, either a-c bus, and any two batteries out of three. The command module and service module structures are still required, but the anticipated margins of safety and internal redundancies make this a highly reliable system. Omitted from Phase III are the command module - reaction control system, earth landing system, and command module - forward heat shield.

The final phase is the entry phase (Figure 8). The basic mode of operation for this phase utilizes the guidance equipment and the automatic control electronics. A spare electronic control assembly is carried. The first alternate mode of operation is through the use of the entry corridor display. The spacecraft can be controlled either through the use of the automatic equipment, including the electronic control assembly and rate gyro package, or in the event of failure, through the use of the manual emergency controls. In either case, the three-axis rotational control must be utilized. The tertiary mode of operation is through the use of communication equipment via the VHF/AM transceiver, utilizing the discone antenna and GOSS. A spare VHF transceiver is provided. The C-band transponder and antenna are required for spacecraft tracking. As shown in previous diagrams, the audio center is also required. Jettisonning of the service module deletes any trajectory correction until atmosphere reentry. Attitude control until drogue chute deployment is provided by redundant halves of the command module-reaction control system. Electrical power is provided by batteries alone, any two out of three.

All elements of command module structure are required for this maneuver of reentry, including the forward heat shield, a single, critical element without redundancy. Earth landing system is redundant internally; i.e., redundant circuits for aft heat shield release mechanism, redundant baro-switches and circuits is the sequencer, a redundant main parachute, and several recovery aids which are redundant among themselves.


INERTIAL MEASUREMENT UNIT
POWER SERVO AMPLIFIER
COUPLING DISPLAY UNIT
APOLLO GUIDANCE COMPUTER
GUIDANCE AND NAVIGATION
ENTRY CORRIDOR DISPLAY
INSTRUMENTATION
VHF AM
VHF AM
DISCONE ANTENNA
C-BAND TRANSPONDER
AUDIO CENTER1
AUDIO CENTER2
AUDIO CENTER3
AUDIO CENTERX
COMMUNICATIONS AND DATA


ELECTRONIC CONTROL ASSEMBLY 1
ELECTRONIC CONTROL ASSEMBLY 2
ELECTRONIC CONTROL ASSEMBLY 3
ELECTRONIC CONTROL ASSEMBLY X
THREE AXIS ROTATIONAL CONTROL
MANUAL EMERGENCY CONTROLS
THREE AXIS ROTATIONAL CONTROL
STABILIZATION AND CONTROL SYSTEM CONTROL PANEL
RATE GYRO PACKAGE
STABILIZATION AND CONTROL
SUIT CIRCUIT
COMPRESSOR
COMPRESSOR
H2O GL
C-BAND ANTENNA
GROUND OPERATIONAL SUPPORT SYSTEM
BATTERY
BATTERY
BATTERY
DC BUSS
DC BUSS
INVERTERS
INVERTERS
INVERTERS
AC BUSS
AC BUSS
ELECTRICAL POWER SYSTEM
HELIUM SUPPLY
OXIDIZER SUPPLY
FUEL SUPPLY
HELIUM SUPPLY
OXIDIZER SUPPLY
FUEL SUPPLY
COMMAND MODULE REACTION CONT

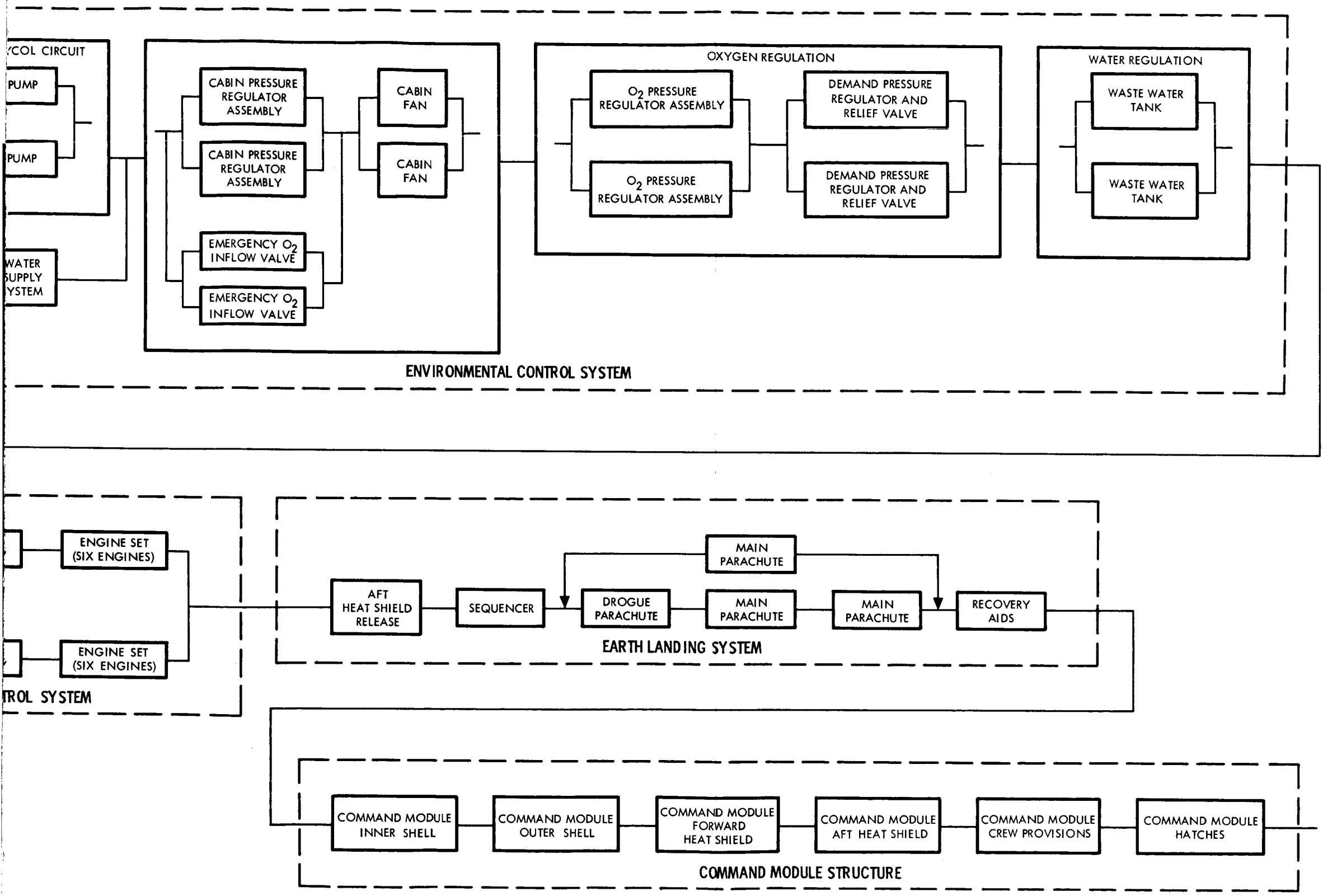


Figure 8. Phase IV - Earth Reentry Through Recovery

ABORT CRITERIA

Table 7 is a list of the Apollo equipments which are construed to be mission essential and which, if they fail, will cause the mission to be aborted. This list is given by individual subsystems. An equipment is considered as having failed only after failure of all redundancy (operational or standby), as well as all spares. The abort criteria is based upon the consideration that the mission will be aborted if sufficient failures occur in operating systems or equipment, so that one additional failure would eliminate the capability of safe abort. This criterion may be modified in individual cases, if it is determined that an unacceptable trade-off between mission success and crew safety reliability will be produced. Examples of equipment to which the above criteria do not apply are the launch escape tower, the sensors, the command module reaction control system, the heat shield, the communication and data subsystem recovery equipment, and the radiation detection equipment.

Table 7. Abort Criteria

Subsystem	Equipments
Guidance and navigation	Inertial measuring unit Power servo assembly Scanning telescope Sextant Coupling display unit Apollo guidance computer
Instrumentation	Central timing equipment
Communications and data	Deep space instrumentation facility transponder C-Band transponder Signal conditioner Audio center plus key 2kmc high-air antenna plus the deep space instrumentation facility power Amplifier or, plus the VHF/2kmc omni-antenna Near earth only, prior to lunar inject: VHF/2kmc omni-antenna Pulse code modulator Premodulation processor VHF/AM transceiver Audio center (3 of 4 units) VHF/FM transmitter or pulse code modulator or premodulation processor plus in-flight test system
Stabilization and control	Electronic control assembly (2 of 4 units) Body mounted attitude gyro Stabilization control subsystem control panel Rate gyro package Three-axis rotational control Flight director attitude indicator Delta V indicator Accelerometer package
Command module structures (0.500 inch hole or greater)	
Service module propulsion (launch to 4th midcourse correction)	
Environmental Control	Suit circuit (1 of 2 units) Water-glycol pumps (1 of 2 units)
Electrical power	Fuel cells (2 of 3 units) Batteries (2 of 3 units) AC bus (1 of 2 units) DC bus (1 of 2 units)
Cryogenic storage	Circuits (1 of 4 units)
Service Module reaction control	Quadrants (1 of 4 units)

METHOD OF ANALYSIS

PROBABILITY DENSITY FUNCTION

This section presents the derivation of two probability density functions required for evaluation of certain Apollo subsystem configurations. The specific subsystems requiring evaluation by use of these functions are classified as being essential to crew safety and include: (1) the environmental control subsystem, (2) the electrical power subsystem, and (3) the cryogenic storage subsystem. These density functions are derived for evaluation of two-element configurations (e.g., fans, compressors, and oxygen supply units) and three element configurations (e.g., inverters and fuel cells) and are based on the abort criteria defined in the previous section.

Definitions

To obtain precise definitions of the reliability terms as they apply to the Apollo Mission, it is necessary to divide the mission into phases consistent with the requirements.

Mission Success

The success for the complete mission is equal to the product of the mission success for each phase. The mission success for each phase (up to transearth inject) is based upon the probability of successful completion of that particular phase without a failure that would require abort. The mission success from transearth inject on is based upon the probability of the return to earth without a failure that would exceed the emergency limits of the crew. In the mathematical models for mission success of each phase, it was necessary to incorporate the conditional probability of survival up to that particular phase.

Mission Failure

The mission failure for each phase up to transearth inject is the probability that the mission will be modified to return to earth prior to its completion. After transearth inject, it is impossible to modify the mission since this is the earth return portion of the mission.

Safe Abort

Safe abort for every phase up to transearth inject is equal to the probability that after mission failure occurs no additional failures will occur that will subject the crew to emergency conditions. After transearth inject, it is impossible to incorporate safe abort since the return-to-earth portion of the mission can not be modified.

Crew Safety

The crew safety for each phase up to transearth inject is defined as the mission success reliability plus the product of the probabilities of mission failure and safe abort for that particular phase of the mission. The crew safety definition from transearth inject to earth return is the same as the mission success definition for that particular phase of the mission. The crew safety for the complete mission is equal to the product of the crew safety reliabilities for each phase of the mission.

Abort Criteria

As a general rule, the mission will be aborted when no mode of operation is available to backup failed equipment.

Mission Phases

For the purpose of mathematical analysis, the mission was divided into four phases.

Phase one - earth launch through earth orbit - 4.7 hrs

Phase two - translunar inject to lunar orbit - 77.17 hrs

Phase three - lunar orbit through LEM stay period one - 64.73 hrs

Phase four - transearth inject through S/M-C/M separation - 189.39 hrs

Assumptions

1. The failure rates were assumed to be constant with respect to time.

2. The units are assumed to have the same failure rate.

3. The units are assumed to be in an operational mode.

4. The failure of any one unit will not affect the function of the remaining units.

5. The assumption was made that each unit would have at least one alternate mode.

Notation

P_{MS} = probability of mission success

Q_{MS} = probability of mission failure

P_{SA} = probability of safe abort

P_{CS} = probability of crew safety

Lack of subscripts designates total mission.

Subscript one is used to designate symbols for Phase I of the mission.

Subscript two is used to designate symbols for Phase II of the mission.

Subscript three is used to designate symbols for Phase III of the mission.

Subscript four is used to designate symbols for Phase IV of the mission.

Mathematical Models for Mission Success and Crew Safety

Using the definitions, assumptions, and notations described herein, the mission success and crew safety was determined for two examples. The requirements for both examples were identical as follows:

Mission Requirements

Mission success

Phase I - two units required

Phase II - two units required

Phase III - two units required

Phase IV - one unit required

Crew safety

One unit required during all phases

To determine the relationship between various time functions due to mission phases, it was necessary to consider the past history of the system. This was accomplished by utilizing the concept of conditional probability. The conditional probability of failure in any time interval T_1 to T_2 can be expressed as follows:

$$F_{T_1 - T_2} = \frac{\int_{T_1}^{T_2} f(t)\, dt}{1 - \int_{0}^{T_1} f(t)\, dt} \qquad (1)$$

where

$F_{T_1 - T_2}$ = Probability of failure in the time interval T_1 to T_2

$f(t)$ = Probability density function of time, t

It should be noted that

$$f(t) = -\frac{dR}{dt}$$

and

$$R(t) = 1 - \int_{0}^{t} f(t)\, dt$$

where

$R(t)$ = The system reliability as a function of time, t.

The conditional probability of survival in the time interval T_2 to T_1 is

$$P_S = 1 - F_{T_1 - T_2} = 1 - \frac{\int_{T_1}^{T_2} f(t)\, dt}{1 - \int_{0}^{T_1} f(t)\, dt} \qquad (2)$$

when R (t) is substituted for $1 - \int_0^{T_1} f(t)\,dt$ in equation (2) the result is

$$P_S = 1 - \frac{\int_{T_1}^{T_2} f(t)\,dt}{R_{T_1}} \qquad (3)$$

The following mathematical models are based upon the application of the type of relationship shown in Equation 3.

EXAMPLE ONE:

Reliability analysis of two units operation in a parallel arrangement.

1. Phase One

(a) Probability of mission success:

$$P_{MS_1} = 1 - \frac{\int_{T_1}^{T_2} f(t)\,dt}{R_{T_1}}$$

R_{T_1} = The probability that the required units will be working at time, T_1. In Phase I T_1 is equal to zero; therefore, RT_1 is equal to unity.

$$f(t) = -\frac{dR}{dt}$$

$$R = e^{-2\lambda t} \quad \therefore \quad f(t) = 2\lambda e^{-2\lambda t}$$

$$P_{MS_1} = 1 - \frac{\int_0^{4.7} f(t)\,dt}{R_{T_0}}$$

since

$$R_{T_0} = 1 \qquad P_{MS_1} = 1 - \int_0^{4.7} 2\lambda\, e^{-2\lambda t}\, dt$$

$$P_{MS_1} = e^{-9.4\lambda} \tag{a}$$

(b) Probability of mission failure:

$$Q_{MS_1} = 1 - P_{MS_1} = 1 - e^{-9.4\lambda}$$

$$Q_{MS_1} = 1 - e^{-9.4\lambda} \tag{b}$$

(c) Probability of safe abort:

It was assumed that if a mission failure occurred in Phase I it would require only one hour to return to earth. Therefore, the remaining unit would have to operate for one hour. Based upon the definitions and assumption;

$$P_{SA_1} = e^{-\lambda} \tag{c}$$

(d) Probability of crew safety:

The probability of crew safety during Phase I can be obtained by using equations (a), (b), and (c).

$$P_{CS_1} = P_{MS_1} + Q_{MS_1} \cdot P_{SA_1}$$

$$P_{CS_1} = e^{-9.4\lambda} + (1 - e^{-9.4\lambda})\, e^{-\lambda}$$

2. Phase Two

(a) Probability of mission success

$$P_{MS_2} = 1 - \frac{\int_{T_2}^{T_3} f(t)\,dt}{R_{T_2}}$$

$$P_{MS_2} = 1 - \frac{\int_{4.7}^{81.87} f(t)\,dt}{R_{4.7}} \qquad f(t) = 2\lambda e^{-2\lambda t}$$

$$P_{MS_2} = 1 - \frac{\int_{4.7}^{81.87} 2\lambda e^{-2\lambda t}\,dt}{e^{-9.4\lambda}}$$

$$= 1 - \left\{ \frac{\left[- e^{-2\lambda t} \right]_{4.7}^{81.87}}{e^{-9.4}} \right\}$$

$$P_{MS_2} = 1 - \left[\frac{- e^{-163.74\lambda} + e^{-9.4\lambda}}{e^{-9.4\lambda}} \right]$$

$$= \frac{e^{-163.74\lambda}}{e^{-9.4\lambda}} \qquad \text{(d)}$$

(b) Probability of mission failure

$$Q_{MS_2} = 1 - P_{MS_2}$$

$$Q_{MS_2} = \left[\frac{e^{-163.74\lambda} - e^{-9.4\lambda}}{e^{-9.4\lambda}} \right] = 1 - \frac{e^{-163.74\lambda}}{e^{-9.4\lambda}}$$

(c) Probability of safe abort

The probability of safe abort during Phase II is dependent on the time, t, at which a failure occurs which necessitates an abort. The time "t" is unknown, therefore, the probability of safe abort is expressed as a function of "t". The maximum time required for an abort in Phase II was assumed to be 154.34, therfore, the time required to abort would be equal to (154.34 - "t"). Utilizing Equation (3) the probability of safe abort is

$$P_{SA_{2(t)}} = 1 - \frac{\int_{t}^{154.34} f(t)\,dt}{R_t}$$

$$R = e^{-\lambda t} \quad \therefore f(t) = \lambda e^{-\lambda t}$$

$$P_{SA_{2(t)}} = 1 - \frac{\int_{t}^{154.34} \lambda e^{-\lambda t}\,dt}{e^{-\lambda t}}$$

$$P_{SA_{2(t)}} = 1 - \left[\frac{-e^{-154.34\lambda} + e^{-\lambda t}}{e^{-\lambda t}}\right] = \frac{e^{-154.34\lambda}}{e^{-\lambda t}}$$

$$P_{SA_2} \cdot Q_{MS_2} = \int_{4.7}^{81.87} \frac{d}{dt} Q_{MS_2}\,dt \int_{0}^{P_{SA_2}} P_{SA_{2(t)}}$$

$$P_{SA_2} \cdot Q_{MS_2} = \int_{4.7}^{81.87} P_{SA_{2(t)}} \frac{dQ_{MS_2}}{dt}\,dt$$

$$= \int_{4.7}^{81.87} \frac{e^{-154.34\lambda}}{e^{-\lambda t}} \left[\frac{2\lambda e^{-\lambda t}}{e^{-9.4\lambda}}\right] dt \qquad (e)$$

[redacted]

(d) Probability of crew safety

The probability of crew safety in Phase II is obtained by using Equations d and e.

$$P_{CS_2} = P_{MS_2} + Q_{MS_2} \cdot P_{SA_2}$$

$$P_{CS_2} = 1 - \left[\frac{e^{-163.74\lambda} - e^{-9.4\lambda}}{e^{-9.4\lambda}}\right] + \int_{4.7}^{81.87} \frac{e^{-154.34\lambda}}{e^{-\lambda t}} \left[\frac{2\lambda e^{-\lambda t}}{e^{-9.4\lambda}}\right] dt$$

3. Phase three

(a) Probability of mission success

$$P_{MS_3} = 1 - \frac{\int_{T_3}^{T_4} f(t)\, dt}{R_{T_3}} \qquad f(t) = 2\lambda e^{-2\lambda t}$$

$$P_{MS_3} = 1 - \frac{\int_{81.87}^{146.6} 2\lambda e^{-2\lambda t}\, dt}{e^{-163.74\lambda}} = \frac{e^{-293.2\lambda}}{e^{-163.74\lambda}}$$

(b) Probability of mission failure

$$Q_{MS_3} = 1 - P_{MS_3} = 1 - \left[\frac{e^{-293.2\lambda}}{e^{-163.74\lambda}}\right]$$

$$Q_{MS_3} = 1 - \left[\frac{e^{-293.2\lambda}}{e^{-163.74\lambda}}\right]$$

[redacted]

[redacted]

(c) Probability of safe abort

It was assumed that if a failure requiring abort occurred in Phase III it would require 65 hours to return to earth. Therefore, the probability of safe abort is a constant in Phase III. The probability of safe abort is

$$P_{SA_3} = e^{-65\lambda}$$

(d) Probability of crew safety

The probability of crew safety in Phase III is:

$$P_{CS_3} = P_{MS_3} + Q_{MS_3} \cdot P_{SA_3}$$

$$P_{CS_3} = \left[\frac{e^{-293.2\lambda}}{e^{-163.74\lambda}}\right] + \left[1 - \left(\frac{e^{-293.2\lambda}}{e^{-163.74\lambda}}\right)\right] e^{-65\lambda}$$

4. Phase four

(a) Probability of mission success

$$P_{MS_4} = 1 - \frac{\int_{T_4}^{T_5} f(t)\, dt}{R_{t_4}}$$

According to the mission requirement only one unit is required for mission success and crew safety, therefore

$$R = e^{-2\lambda t} + 2e^{-\lambda t}(1 - e^{-\lambda t})$$

$$R = e^{-2\lambda t} + 2e^{-\lambda t} - 2e^{-2\lambda t}$$

$$R = 2e^{-\lambda t} - e^{-2\lambda t}$$

$$f(t) = -\frac{dR}{dt} = -[-2\lambda e^{-\lambda t} + 2\lambda e^{-2\lambda t}]$$

$$f(t) = 2\lambda e^{-\lambda t} - 2\lambda e^{-2\lambda t}$$

$$R_{t_4} = 2e^{-146.6\lambda} - e^{-293.2\lambda}$$

$$P_{MS_4} = 1 - \frac{\int_{146.6}^{335.94} (2\lambda e^{-\lambda t} - 2\lambda e^{-2\lambda t})\, dt}{2e^{-146.6\lambda} - e^{-293.2\lambda}}$$

$$P_{MS_4} = 1 - \left\{ \frac{\left[-2e^{-\lambda t} \right]_{146.6}^{335.94}}{2e^{-146.6\lambda}} - \frac{\left[-e^{-2\lambda t} \right]_{146.6}^{335.94}}{e^{-293.2\lambda}} \right\}$$

(b) Probability of crew safety

According to the definitions stated within, the probability of crew safety in Phase IV is equal to the probability of mission success in Phase IV.

$$P_{CS_4} = P_{MS_4}$$

5. Over-all mission success

The probability of mission success for the complete mission is

$$P_{MS} = P_{MS_1} \cdot P_{MS_2} \cdot P_{MS_3} \cdot P_{MS_4}$$

6. Over-all crew safety

The probability of crew safety for the complete mission is

$$P_{CS} = \left[\left(P_{MS_1} + Q_{MS_1} \cdot P_{SA_1} \right) \cdot \left(P_{MS_2} + Q_{MS_2} \cdot P_{SA_2} \right) \cdot \left(P_{MS_3} + Q_{MS_3} \cdot P_{SA_3} \right) \right] P_{MS_4}$$

$$P_{CS} = P_{CS_1} \cdot P_{CS_2} \cdot P_{CS_3} \cdot P_{CS_4}$$

EXAMPLE TWO:

Reliability analysis of three units operating in a parallel arrangement.

1. Phase one

(a) Probability of mission success

$$P_{MS_1} = 1 - \frac{\int_{T_0}^{T_1} f(t)\,dt}{R_{T_0}}$$

$$R = e^{-3\lambda t} + 3e^{-2\lambda t}(1 - e^{-\lambda t}) =$$

$$-2e^{-3\lambda t} + 3e^{-2t}$$

$$f(t) = -\frac{dR}{dt} = -(6\lambda e^{-3\lambda t} - 6\lambda e^{-2\lambda t})$$

R_{T_0} = unity for phase one

$$P_{MS_1} = 1 - \int_0^{4.7} (-6\lambda e^{-3\lambda t} + 6\lambda e^{-2\lambda t})\,dt$$

$$P_{MS_1} = 1 - [2e^{-3\lambda t} - 3e^{-2\lambda t}]_0^{4.7}$$

(b) Probability of mission failure

$$Q_{MS_1} = 1 - P_{MS_1}$$

$$Q_{MS_1} = [2e^{-3\lambda t} - 3e^{-2\lambda t}]_0^{4.7}$$

(c) Probability of safe abort

The assumption stated in Example One concerning an abort in Phase I was applied

$$P_{SA_1} = e^{-\lambda}$$

(d) Probability of crew safety

The probability of crew safety in Phase I is

$$P_{CS_1} = P_{MS_1} + Q_{MS_1} \cdot P_{SA_1}$$

$$P_{CS_1} = \left\{ 1 - \left[2e^{-3\lambda t} - 3e^{-2\lambda t} \right]_0^{4.7} \right\} + \left[2e^{-3\lambda t} - 3e^{-2\lambda t} \right]_0^{4.7} \cdot e^{-\lambda}$$

2. Phase two

(a) Probability of mission success

$$P_{MS_2} = 1 - \frac{\int_{T_1}^{T_2} f(t)\, dt}{R_{T_1}}$$

$$R = -2e^{-3\lambda t} + 3e^{-2\lambda t}$$

$$f(t) = -\frac{dR}{dt} = -6\lambda e^{-3\lambda t} + 6\lambda e^{-2\lambda t}$$

$$R_{4.7} = -2e^{-14.1\lambda} + 3e^{-9.4\lambda}$$

$$P_{MS_2} = 1 - \frac{\int_{4.7}^{81.8} -(6\lambda e^{-3\lambda t} - 6\lambda e^{-2\lambda t})\, dt}{-2e^{-14.1\lambda} + 3e^{-9.4\lambda}}$$

$$P_{MS_2} = 1 - \left\{ \frac{\left[2 e^{-3\lambda t} \right]_{4.7}^{81.8} + \left[-3 e^{-2\lambda t} \right]_{4.7}^{81.8}}{-2 e^{-14.1\lambda} + 3 e^{-9.4\lambda}} \right\}$$

(b) Probability of mission failure

$$Q_{MS_1} = 1 - P_{MS_2}$$

$$Q_{MS_2} = \frac{\left[2 e^{-3\lambda t} \right]_{4.7}^{81.8} + \left[-3 e^{-2\lambda t} \right]_{4.7}^{81.8}}{-2 e^{-14.1\lambda} + 3 e^{-9.4\lambda}}$$

(c) Probability of safe abort

As stated in Example One, the probability of safe abort in Phase II is a function of time.

$$P_{SA_{2(t)}} = 1 - \frac{\int_t^{154.34} f(t)\, dt}{R_t}$$

$$R = e^{-\lambda t} \quad \therefore f(t) = \lambda e^{-\lambda t}$$

$$P_{SA_{2(t)}} = 1 - \frac{\int_t^{154.34} \lambda e^{-\lambda t}\, dt}{e^{-\lambda t}}$$

$$P_{SA_{2(t)}} = \frac{e^{-154.34\lambda}}{e^{-\lambda t}}$$

$$P_{SA_2} \cdot Q_{MS_2} = \int_{4.7}^{81.87} \frac{d\, QMS_2\, dt}{dt} \cdot \int_0^{P_{SA_2}} P_{SA_{2(t)}}$$

$$P_{SA_2} \cdot Q_{MS_2} = \int_{4.7}^{81.87} P_{SA_{2(t)}} \frac{d\,Q_{MS_2}}{dt}\, dt$$

$$P_{SA_2} \cdot Q_{MS_2} = \int_{4.7}^{81.87} \frac{e^{-154.34\lambda}}{e^{-\lambda t}} \left[\frac{-6\lambda e^{-3\lambda t} + 6\lambda e^{-2\lambda t}\, dt}{-2e^{-14.1\lambda} + 3e^{-9.4\lambda}} \right]$$

(d) Probability of crew safety

The probability of crew safety is;

$$P_{CS_2} = P_{MS_2} + Q_{MS_2} P_{SA_2}$$

3. Phase three

(a) Probability of mission success

$$P_{MS_3} = 1 - \frac{\int_{T_2}^{T_3} f(t)\, dt}{R_{T_2}}$$

$$R = -2e^{-3\lambda t} + 3e^{-2\lambda t}$$

$$f(t) = -6\lambda e^{-3\lambda t} + 6\lambda e^{-2\lambda t}$$

$$R_{T_2} = -2e^{-245.61\lambda} + 3e^{-163.74\lambda}$$

$$P_{MS_3} = 1 - \frac{\int_{81.87}^{146.6} (-6\lambda e^{-3\lambda t} + 6\lambda e^{-2\lambda t})\, dt}{-2e^{-245.61\lambda} + 3e^{-163.74\lambda}}$$

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$$P_{MS_3} = 1 - \frac{\left[2e^{-3\lambda t}\right]_{81.87}^{146.61} + \left[-3e^{-2\lambda t}\right]_{81.87}^{146.61}}{-2e^{-245.61\lambda} + 3e^{-163.74\lambda}}$$

(b) Probability of mission failure

$$Q_{MS_3} = 1 - P_{MS_3}$$

$$Q_{MS_3} = \frac{\left[2e^{-3\lambda t}\right]_{81.87}^{146.61} + \left[-3e^{-2\lambda t}\right]_{81.87}^{146.61}}{-2e^{-245.61\lambda} + 3e^{-163.74\lambda}}$$

(c) Probability of safe abort

As stated in Example One, the probability of safe abort in Phase III is a constant

$$P_{SA_3} = e^{-65\lambda}$$

(d) Probability of crew safety

$$P_{CS_3} = P_{MS_3} + Q_{MS_3} \cdot P_{SA_3}$$

4. Phase four

(a) Probability of mission success

$$P_{MS_4} = 1 - \frac{\int_{T_3}^{T_4} f(t)\,dt}{R_{T_3}}$$

$$R = e^{-3\lambda t} + 3e^{-2\lambda t}(1 - e^{-\lambda t}) + 3e^{-2\lambda t}(1 - e^{-\lambda t})^2$$

$$R = e^{-3\lambda t} - 3e^{-2\lambda t} + 3e^{-\lambda t}$$

$$f(t) = -\frac{dR}{dt} = 3\lambda e^{-3\lambda t} - 6\lambda e^{-2\lambda t} + 3\lambda e^{-\lambda t}$$

$$R_{T_3} = e^{-439.8\lambda} - 3e^{-293.2\lambda} + 3e^{-146.6\lambda}$$

$$P_{MS_4} = 1 - \frac{\int_{146.6}^{335.94} (3\lambda e^{-3\lambda t} - 6\lambda e^{-2\lambda t} + 3\lambda e^{-\lambda t})\, dt}{e^{-439.8\lambda} - 3e^{-293.2\lambda} + 3e^{-146.6\lambda}}$$

$$P_{MS_4} = 1 - \frac{\left[-e^{-3\lambda t} + 3e^{-2\lambda t} - 3e^{-\lambda t}\right]_{146.6}^{335.94}}{e^{-439.8\lambda} - 3e^{-293.2\lambda} + 3e^{-146.6\lambda}}$$

(b) Probability of crew safety

$$P_{CS_4} = P_{MS_4}$$

5. Over-all mission success

The probability of mission success for the complete mission is

$$P_{MS} = P_{MS_1} \cdot P_{MS_2} \cdot P_{MS_3} \cdot P_{MS_4}$$

6. Over-all crew safety

The probability of crew safety for the complete mission is;

$$P_{CS} = \left[\left(P_{MS_1} + Q_{MS_1} \cdot P_{SA_1}\right) \cdot \left(P_{MS_2} + Q_{MS_2} \cdot P_{SA_2}\right) \cdot \left(P_{MS_3} + Q_{MS_3} P_{SA_3}\right)\right] \cdot P_{MS_4}$$

$$P_{CS} = P_{CS_1} \cdot P_{CS_2} \cdot P_{CS_3} \cdot P_{CS_4}$$

[redacted]

MECHANICAL SUBSYSTEMS

This section contains a description of all mechanical subsystems within the spacecraft and the various components within the subsystems. The reliability estimates for these individual components and subsystems are of secondary importance; the total spacecraft reliability, described in a later section that includes the backup of components within one subsystem by components in another, gives a more important evaluation of the subsystem.

CRYOGENIC STORAGE SUBSYSTEM

Accomplishments on the cryogenic storage subsystem during the past quarter include completion of reliability predictions for the present system configuration, preparation of a detailed failure effects analysis, and an evaluation of a simplified system configuration proposed by design engineering for increasing the inherent reliability of the system. Table 8 summarizes the results of this analysis and compares the reliability of the two configurations with the requirements.

Table 8. Reliability Analysis Results

Configuration	Mission Success	Crew Safety
Present design	0.9946	0.99977
Simplified design	0.9965	0.99996
Requirement	0.9989	0.99999

Future activity planned for analysis of this subsystem will include a concerted effort with the design group to further simplify the system design. A detailed failure effects analysis of the modified configuration will be prepared.

CONFIGURATION ANALYSIS

Subsystem Description

The cryogenic storage subsystem is used to provide reactant supply to the electrical power subsystem fuel cells and oxygen supply to the environmental control subsystem. The components of the subsystem are listed in Table 9.

Table 9. Cryogenic Storage Subsystem Components

Oxygen		Hydrogen	
Function	Symbol	Function	Symbol
Supply (includes distribution)	R_1	Supply (includes distribution)	R_6
Purge	R_2	Purge	R_7
Fittings	R_3	Fittings	R_8
Lines	R_4	Lines	R_9
Controls	R_5	Controls	R_{10}

Abort Criteria

The mission shall be aborted any time a failure occurs of such a nature that a subsequent failure would mean the loss of the crew.

Present Design

A reliability analysis of the present system configuration was performed to evaluate its capability to meet reliability requirements. Concurrently, a failure effects analysis of the design was completed. The results of this analysis are shown in Table 12 at the end of this section. This analysis studied the possible modes of failure of the system and its components. The analysis revealed possible solutions for the various problems which may occur as a result of one or more failures. The results of this analysis form the logic for the block diagrams and math models included in this report. Results of the analysis indicated that the present system is too complex, thereby compromising system reliability. Figure 9 presents a schematic of this subsystem.

Mission Success

The total probability of success for the cryogenic storage subsystem is determined by calculating the probability that no failure(s) will occur which would require the mission to be aborted during the first 146 hours and the probability of safe return during the last 190 hours. (This last probability is identical to the logic of crew safety.) Figure 10 presents a logic block diagram for the first 146 hours of the mission. Although Figure 11 presents a logic block diagram for crew safety for 336 hours, it is also representative of mission success for the last 190 hours.

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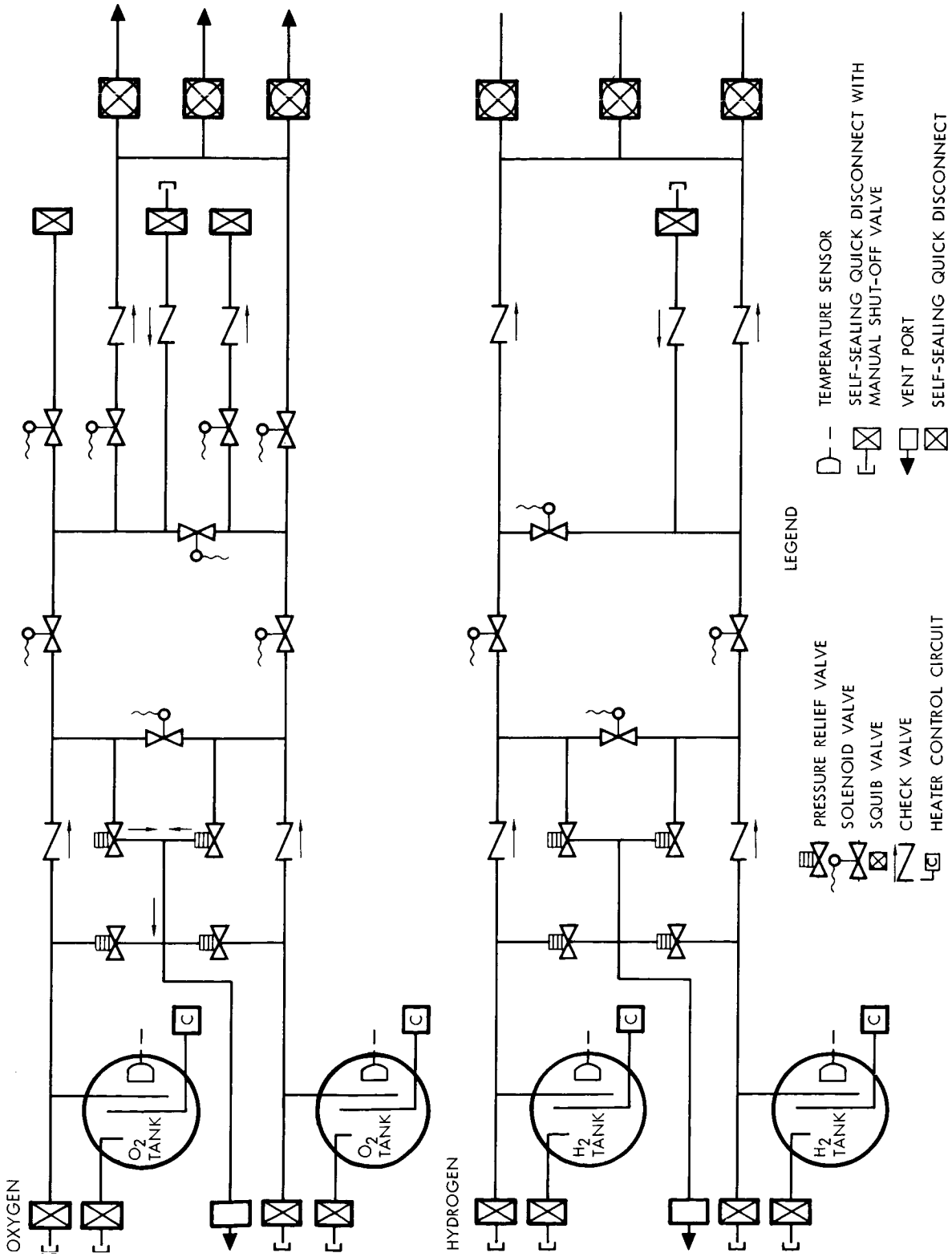

Figure 9. Cryogenic Storage Subsystem Present Design

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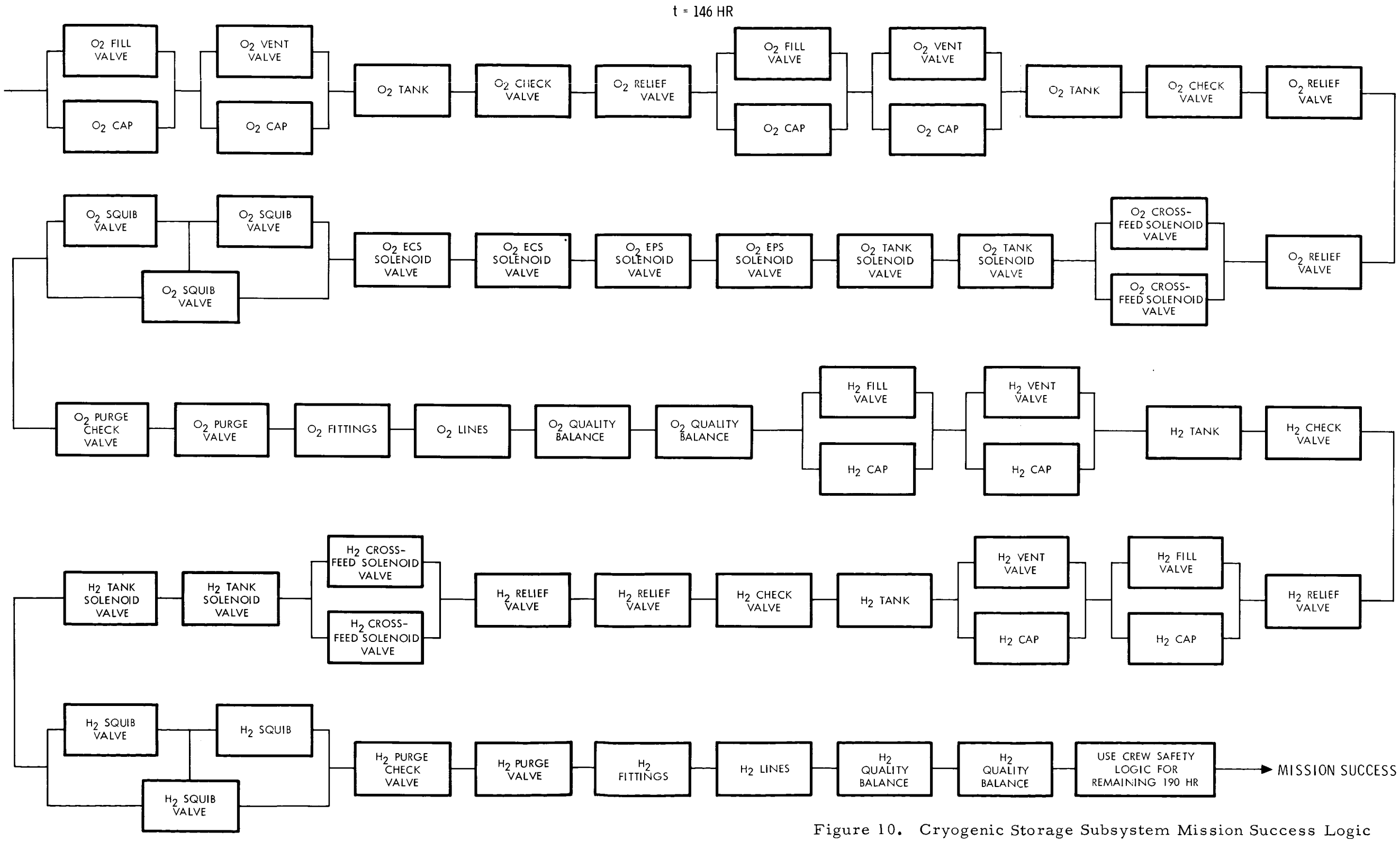
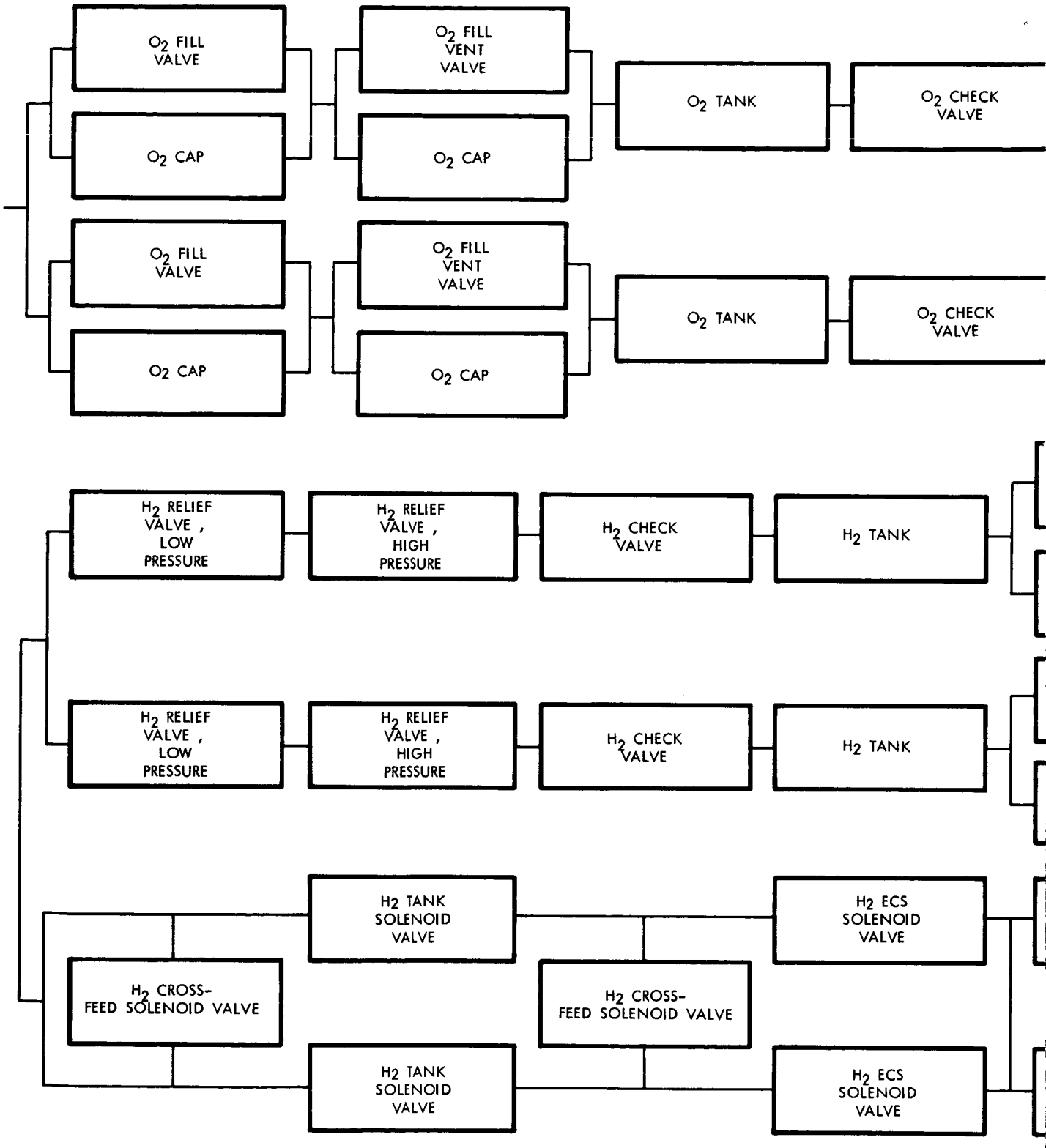


Figure 10. Cryogenic Storage Subsystem Mission Success Logic Diagram - Present Design


O_2 FILL VALVE
O_2 CAP
O_2 FILL VENT VALVE
O_2 CAP
O_2 TANK
O_2 CHECK VALVE
O_2 FILL VALVE
O_2 CAP
O_2 FILL VENT VALVE
O_2 CAP
O_2 TANK
O_2 CHECK VALVE
H_2 RELIEF VALVE, LOW PRESSURE
H_2 RELIEF VALVE, HIGH PRESSURE
H_2 CHECK VALVE
H_2 TANK
H_2 RELIEF VALVE, LOW PRESSURE
H_2 RELIEF VALVE, HIGH PRESSURE
H_2 CHECK VALVE
H_2 TANK
H_2 TANK SOLENOID VALVE
H_2 ECS SOLENOID VALVE
H_2 CROSS-FEED SOLENOID VALVE
H_2 CROSS-FEED SOLENOID VALVE
H_2 TANK SOLENOID VALVE
H_2 ECS SOLENOID VALVE

t = 336 HR

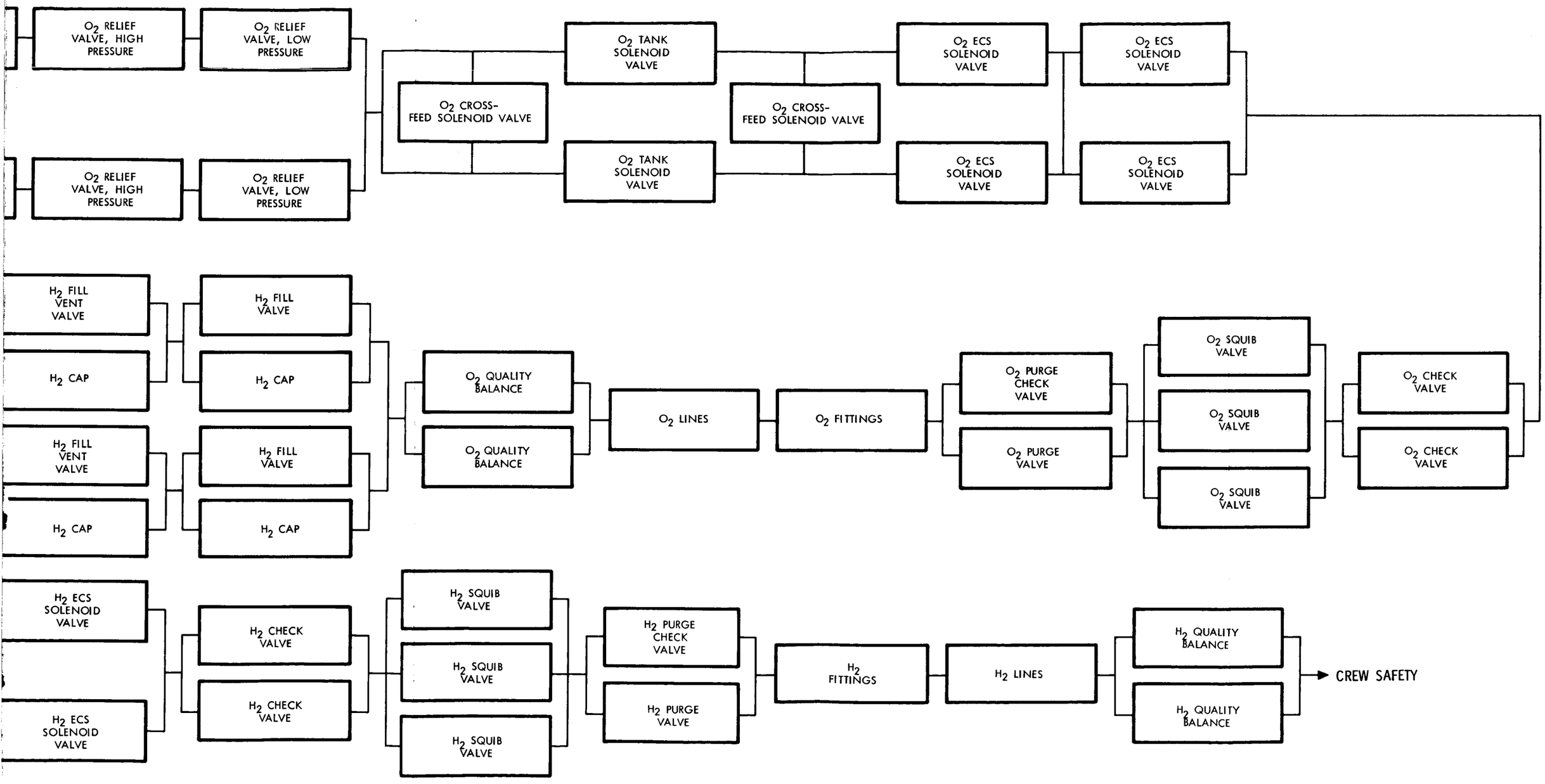


Figure 11. Cryogenic Storage System Crew Safety Logic Diagram - Present Design

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System Equation

$$R_{system} = R^2_{oxygen} \left\{ 1 - \left(1 - R_{oxygen_2}\right)^2 \right\} R^2_{hyd} \left\{ 1 - \left(1 - R_{hyd_2}\right)^2 \right\}$$

where:

R_{oxygen} = $R_1 \times R_2 \times R_3 \times R_4 \times R_5$ (see Table 9)

R_{oxygen_1} = Reliability of oxygen subsystem for 146 hours

R_{oxygen_2} = Reliability of oxygen subsystem for 190 hours

$R_{hydrogen}$ = $R_6 \times R_7 \times R_8 \times R_9 \times R_{10}$ (see Table 9)

$R_{hydrogen_1}$ = Reliability of hydrogen system for 146 hours

$R_{hydrogen_2}$ = Reliability of hydrogen system for 190 hours

R_{system} = 5356 aborts/million missions (figures based on data from Table 10)

Crew Safety

The safety of the crew during the mission is determined by calculating the probability that a sufficient supply of reactants to the fuel cells and a sufficient supply of oxygen to the crew will be provided throughout the mission. Figure 11 presents a logic diagram for the crew safety probability.

System Equation

Equations used in evaluating the crew safety are based on the probability density function derivation.

R_{oxygen} = $R_1 \times R_2 \times R_3 \times R_4 \times R_5$ (See Table 9)

R_{oxygen} = Reliability of the oxygen subsystem for 336 hours

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Table 10. Cryogenic Storage Component Failure Rates and Reliability Values

Component	Failure Rate $\times 10^{-6}$	t.Fr $\times 10^{-6}$ (t=336 hr)	t.Fr $\times 10^{-6}$ (t=190 hr)	t.Fr $\times 10^{-6}$ (t=146 hr)	"R" (t = 336 hr)	"R" (t = 190 hr)	"R" (t = 146 hr)
Tank	0.078	26.2	1.5	1.1	0.999974	0.999999	0.999999
Solenoid valve	1.06	356.2	201.4	154.8	0.999644	0.999799	0.999845
Relief valve	0.96	322.6	182.4	140.2	0.999677	0.999818	0.999860
Check valve	1.73	581.3	328,7	252.6	0.999419	0.999671	0.999747
Quick disconnect	0.39	131.0	74.1	56.9	0.999869	0.999926	0.999943
Manual closure	0.00077	0.3	0.1	0.1	0.9999997	0.9999999	0.9999999
Squib valve	0.0008	0.3	0.1	0.1	0.9999997	0.9999999	0.9999999
Quantity balance	0.466	156.6	88.5	68.0	0.999843	0.999912	0.999932
Fittings	0.00077	0.3	0.1	0.1	0.9999997	0.9999999	0.9999999
Lines	0.0017	0.6	0.3	0.2	0.9999994	0.9999997	0.9999998

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$R_{hydrogen} = R_6 \times R_7 \times R_8 \times R_9 \times R_{10}$ (See Table 9)

$R_{hydrogen}$ = Reliability of the hydrogen subsystem for 336 hours

R_{system} = 232 failures/million missions (Figures based on data from Table 11)

Simplified Subsystem Design

A reliability analysis was performed to determine what effect simplification of the subject system would have on crew safety and on mission abort probability. The simplified design schematic is shown in Figure 12.

As a result of this analysis, it has been determined that a significant increase in system reliability has been effected by the efforts to simplify system design; however, the requirements for mission success and crew safety have not as yet been met. Table 11 presents the results of the analysis and compares the estimates with the requirements.

Table 11. Results of Reliability Analysis for Crew Safety and Mission Success

Results of Configuration Analysis	Crew Safety	Abort Probability
Present design	232 failures per million missions	5356 aborts per million missions
Simplified design	43 failures per million missions	3504 aborts per million missions
Reliability objective (No greater than)	No greater than 10 failures per million missions	No greater than 1100 aborts per million missions
Note: Figures based on state-of-the-art data		

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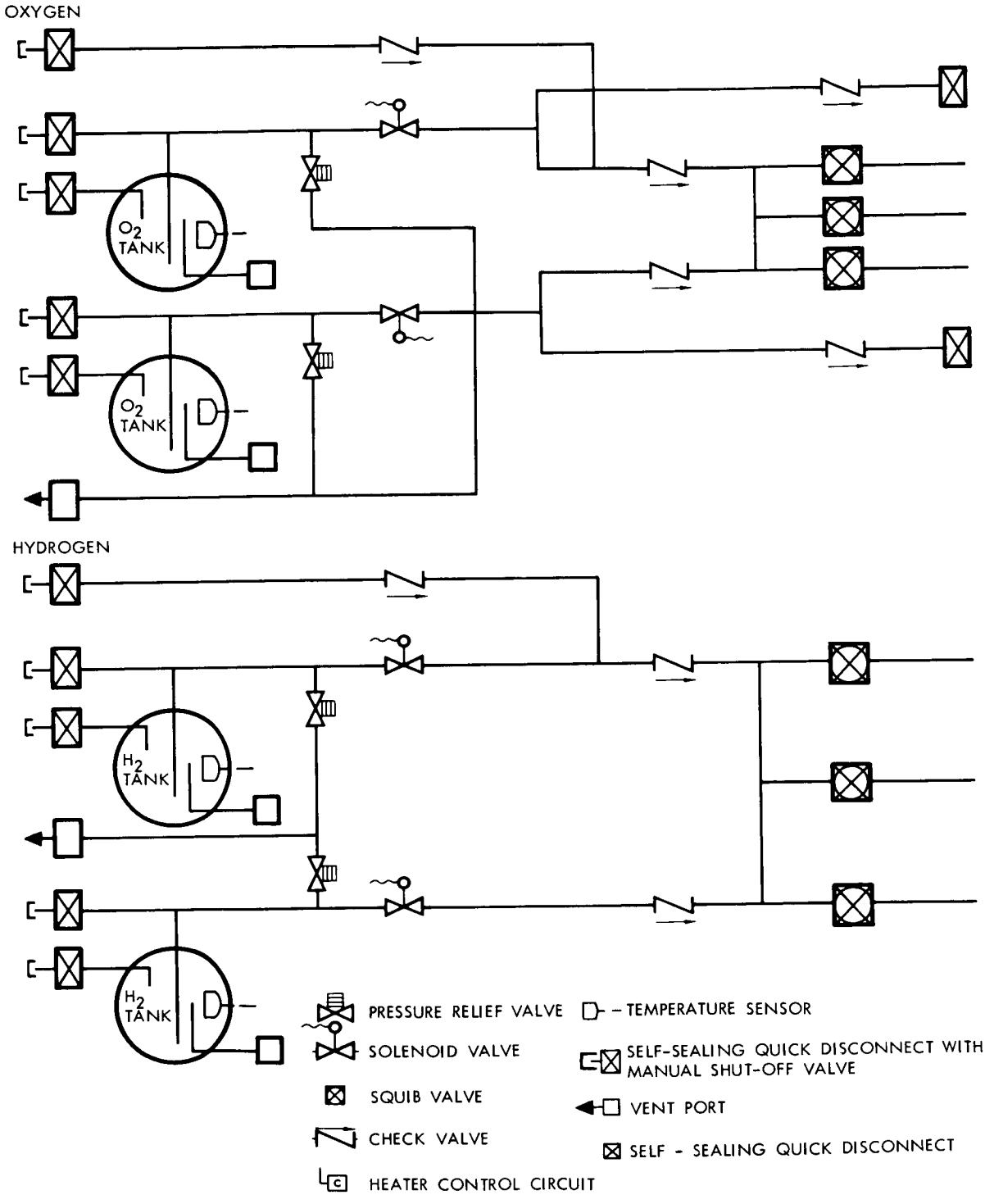

Figure 12. Cryogenic Storage Subsystem Simplified Design

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Mission Success

The mission success probability for the cryogenic storage subsystem is determined by calculating the probability that no failure(s) will occur that would require the mission to be aborted during the first 146 hours and the probability of safe return during the last 190 hours (the latter probability logic is identical to the logic for crew safety). Figure 13 presents a logic diagram for the remaining 190 hours of the mission success probability.

System Equation

$$R_{system} = R^2_{oxygen_1} \left[1 - \left(1 - R_{oxygen_2}\right)^2\right] R^2_{hydrogen_1} \left[1 - \left(1 - R_{hydrogen_2}\right)^2\right]$$

where:

$R_{oxygen} = R_1 \times R_2 \times R_3 \times R_4 \times R_5$ (See Table 9)

R_{oxygen_1} = Reliability of oxygen subsystem for 146 hours

R_{oxygen_2} = Reliability of oxygen subsystem for 190 hours

$R_{hydrogen} = R_6 \times R_7 \times R_8 \times R_9 \times R_{10}$ (See Table 9)

$R_{hydrogen_1}$ = Reliability of hydrogen subsystem for 146 hours

$R_{hydrogen_2}$ = Reliability of hydrogen subsystem for 190 hours

R_{system} = 3504 aborts/million missions (Figures based on data from Table 10.)

Crew Safety

The safety of the crew during the mission is determined by calculating the probability that a sufficient supply of reactants to the fuel cells and a sufficient supply of oxygen to the crew will be provided throughout the mission. Figure 14 presents a logic diagram for the crew safety probability.

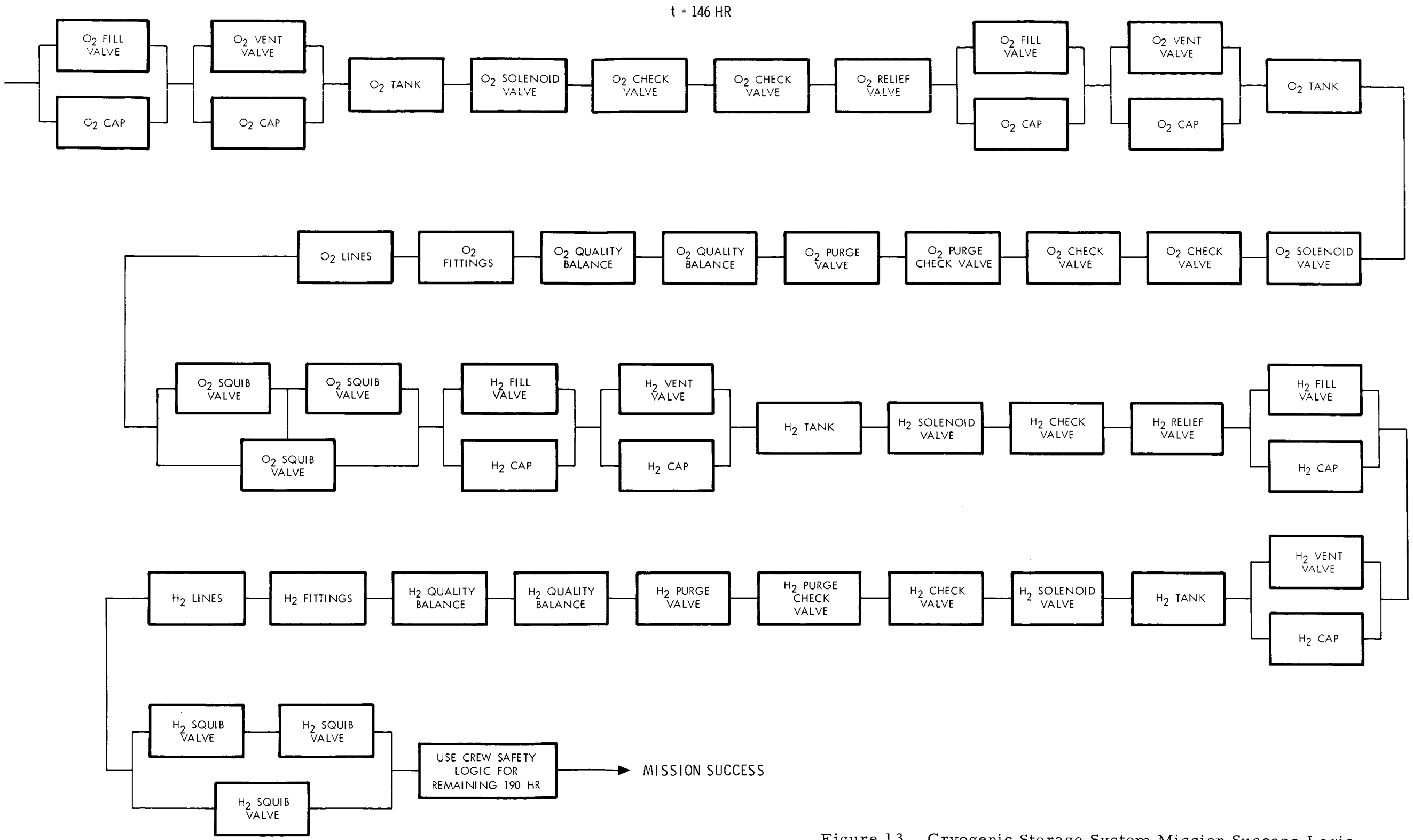

Figure 13. Cryogenic Storage System Mission Success Logic Diagram - Simplified System

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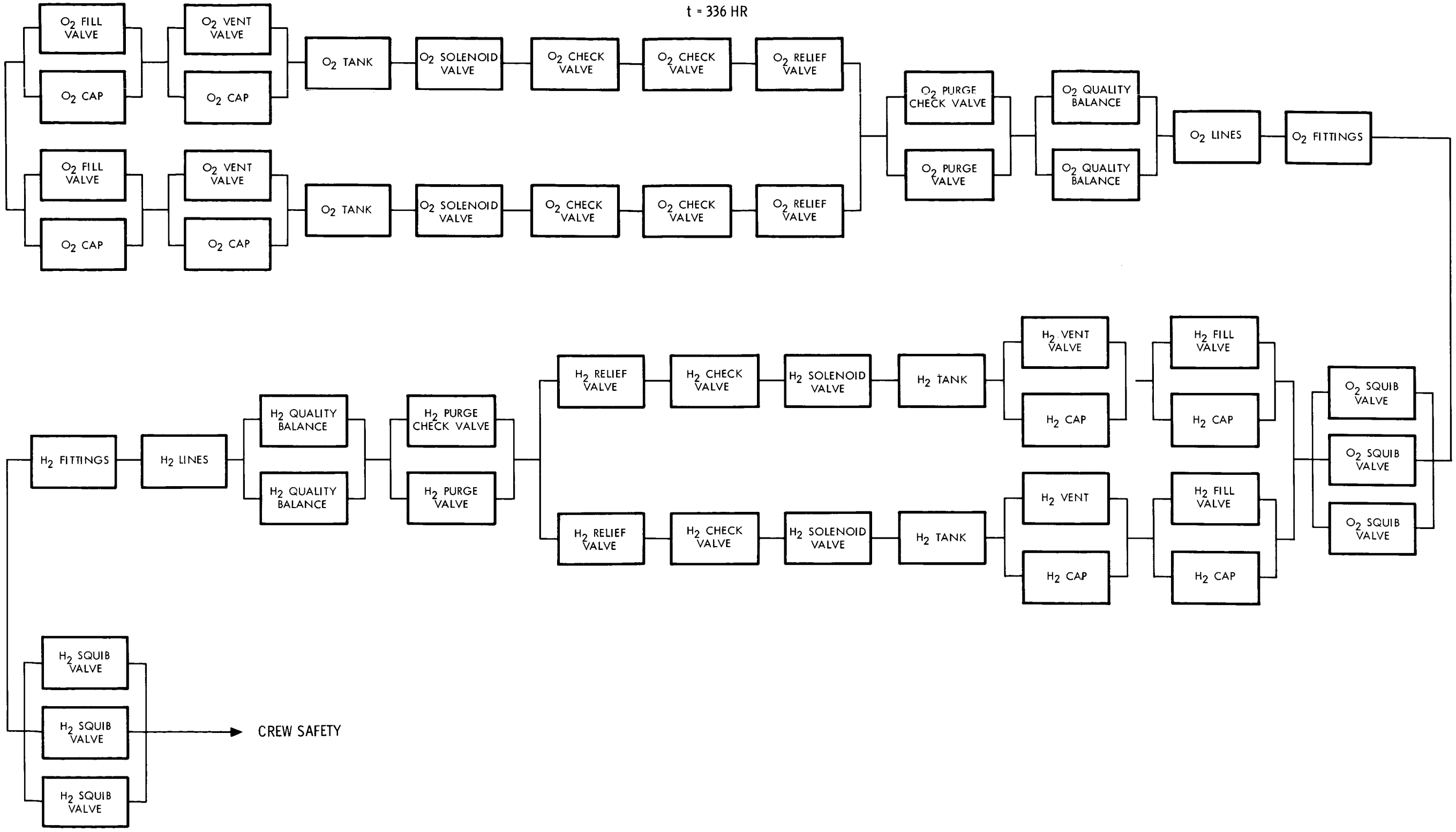


Figure 14. Cryogenic Storage System Crew Safety Logic Diagram - Simplified System

System Equation

Equations used in evaluating the crew safety are based on the density function derivation shown in the Subsystem Method of Analysis section.

Where:

$R_{oxygen} = R_1 \times R_2 \times R_3 \times R_4 \times R_5$

R_{oxygen} = Reliability of the oxygen subsystem for 336 hours

$R_{hydrogen} = R_6 \times R_7 \times R_8 \times R_9 \times R_{10}$

$R_{hydrogen}$ = Reliability of the hydrogen subsystem for 336 hours

R_{system} = 43 Failures/Million Missions (Figures based on data from Table 10.)

The present design has a crossfeed distribution system which uses twelve solenoid valves, ten check valves, and eight relief valves. The simplified design uses four solenoid valves, six check valves, and four relief valves. (See Figures 9, 10, 11.) In addition to being less complex, the simplified system has a higher crew safety reliability and lower mission abort probability. The solenoid valves in the crossfeed are considered necessary to protect the system from leakage due to a rupture. Since a leak in one line would require a mission abort and since either of the storage tanks holds enough fluid to return the crew from any point in the mission, there seems to be no advantage in the extra valves.

The simplified design has the advantage of being lower in weight as a result of the removal of fourteen components. The simplified system will cost less to develop and produce because of the use of fewer components and the replacement of low-reliability components with high-reliability components.

Since a manual closure has a high reliability, there is no basis, from a reliability standpoint, for a check valve being in the purge line. If there is no basis for the check valve installation from a design standpoint, Reliability Engineering recommends the removal of the valve.

Considering the factors of reliability and safety, Reliability Engineering recommended that a re-evaluation of the cryogenic storage system be initiated by the design group with a view towards simplification.

TEST PROGRAM

Qualification Test Plan

Qualification testing has been revised to reflect the new test philosophy and the latest environmental criteria.

Two coordination meetings were attended during this reporting period. The first was held at S&ID in November, 1962, to clarify the specification requirements. The second, held at S&ID in December, 1962, was to clarify the new qualification test philosophy.

General Test Program Status

Following is the development test status of the cryogenic storage system.

1. Proof pressure tests on the titanium tanks were successfully completed.

2. Test procedures for the titanium tank LH_2 cryogenic proof and burst tests were completed.

3. Test procedures for the titanium tank volumetric expansion at cryogenic temperatures test were completed.

4. Vibration tests on the vacuum relief valve are now in progress.

5. Insulation segment test procedures were completed.

Effects of Qualification Program Reorientation

Better utilization of the allotted test specimens to achieve a more realistic test program was accomplished as a result of qualification program redirection.

SUBCONTRACTOR MANAGEMENT

Subcontractor Coordination

More emphasis is being put on coordination efforts between Beech Cryogenics and S&ID Reliability Departments. With the advent of contract bidding by Beech subcontractors, substantial changes in the

NAA specification on the subject system, and a new test philosophy, coordination has become a major area of interest. Efforts coordinated thus far include the following:

1. Beech requested IDEP information and S&ID sponsored Beech's entry into the program.

2. Revision of the NAA specification on the subject system was a direct result of the October coordination meeting. The reliability portion was revised to specify reliability requirements and definitions.

3. A new statistical approach to testing was agreed upon.

4. Assignment of system responsibilities and definitions of various levels of systems, subsystems, and components were agreed upon.

5. A single-line system for use in the simplified system was considered. The single line is not feasible because it would require moving the system check valves close to the source of gas, thereby causing excessive cycling and wear of the valve.

6. It is desirable to locate the ECS check valves in the command module to expose the valves to the best possible operating environment.

7. Separate reliability testing will not be performed. Reliability calculations will be based on the results of development and qualification tests.

8. To give better direction and orientation to subcontractor reliability programs, a publication is being prepared by S&ID Reliability which outlines the reliability program for Apollo subcontractors. Beech is to receive a complete set of these documents upon publication.

Subcontractor Effort

Test procedures for the insulation segment test and tooling have been completed. Testing has been started on the vacuum relief valve vibration requirements. Procedures have also been established and are in the process of approval for the fill and vent coil testing, oxygen heater vibration, engineering model shock and vibration, and strain gauge evaluation of pressure vessels.

Proof pressure tests of the airtight titanium vessel have been completed, and procedures for LH_2 cryogenic proof and burst tests are in progress. Test procedures for volumetric expansion at cryogenic temperatures is in process.

Reliability analysis data on the oxygen system was transmitted to S&ID by Beech representatives at a meeting held at the NAA-Downey facility. The data included the following:

1. Schematic of the simplified design
2. Schematic of the present design
3. Block diagrams of the simplified and the present design
4. Failure rate apportionment for both configurations
5. Reliability study using R = 0.9989 for each of four units, compared to a study using R = 0.9989 for the result of a series of the four units. This study included degrading effects on all units.

The analysis did not include a calculation of mission success using the 154-hour requirement. This analysis will be transmitted to S&ID by the next Beech monthly report.

A preliminary failure mode analysis submitted by Beech is now being reviewed by Apollo Reliability.

PLANNED ACTIVITIES

With the acceptance by NASA of the simplified design, work in the next quarterly reporting period should include the following:

1. Failure effects analysis of the simplified design
2. Failure rate apportionment for components, as used in the simplified design
3. Studies of system and component degrading during normal mission
4. Complete system reliability analysis by subcontractors
5. Initial results of testing now in progress

Table 12. Cryogenic Storage Subsystem Failure Effects Analysis

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Fill valve	Beech 13532-4005		Allows filling and prevents leakage during mission	Leakage	Launch to service module separation	Erosion of seal	Loss of gas in amount exceeding allowable leakage	Insufficient gas supply to complete the 336 hour mission	Detect leak, abort mission, close off good supply, use as much of failed system as possible, then close failed system and switch to good system for abort completion.
Vent valve	Beech 13532-4005		Allows venting during fill operation and prevents leakage thereafter	Leakage	Launch to service module separation	Erosion of seal	Loss of gas in amount exceeding allowable leakage	Insufficient gas to complete the 336 hour mission	Detect leak, abort mission, close off good supply, use as much of failed system as possible, then close failed system and switch to good system for abort completion.
Supply check valve	Beech 13532-4005		Prevents loss of gas through a leak in the system	Leakage	Launch to service module separation	Erosion of seal	Loss of gas in amount exceeding allowable leakage	Insufficient gas to complete the 336 hour mission	Close supply valve.
Supply relief valve	Beech 13532-4005		Prevents over pressurization and seals itself against leakage when reseating	Leakage	Launch to service module separation	Erosion of seal	Loss of gas in amount exceeding allowable limit	Insufficient gas to complete the 336 hour mission	Close supply valve.
Supply shut-off valve (solenoid latch type)	Beech 13532-4006		Shuts off gas supply from system for purge operation and tank isolation	Inadvertantly closes or actuates solenoid switch	Launch to service module separation	None	Leaves only one path for gas to crew and power cells	Failure of the other supply valve could now cause loss of crew	Throw solenoid switch to OPEN position. If this fails, abort mission. Use cross-feed system to utilize failed system.

Table 12. Cryogenic Storage Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Supply shut-off valve (solenoid latch type) (Cont)				Leakage	Launch to service module separation	Erosion of seals	Loss of gas in amount exceeding allowable limit	Insufficient gas to complete the 336 hour mission	Detect leak, abort mission, close good supply, use as much of failed system as possible, then close failed system and switch to good system for abort completion.
Redundant supply shut-off valve (solenoid latch type)	Beech 13532-4006		Closes supply line in the event of leakage	Inadvertantly closes	Launch to service module separation	None	Leaves only one path for gas to crew and power cells	Failure of parallel supply line valve could now cause loss of crew	Throw solenoid switch to OPEN position. If this fails, abort mission. Use crossfeed system to utilize failed system.
				Leakage	Launch to service module separation	Erosion of seals	Loss of gas in amount exceeding allowable limit	Insufficient gas to complete the 336 hour mission	Detect leak, abort mission, close good supply, use as much of failed system as possible, then close failed system and switch to good system for abort completion.
Quantity and pressure control	Beech 13532-0030		Maintains system pressure and ensures use of equal quantities of fluid from both tanks by controlling the amount of heat applied to the fluid in the tanks	Open circuit (wiring)	Launch to service module separation	None	Pressure loss in one of the tanks and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance.
				Pressure switch fails open	Launch to service module separation	Component rendered useless	Pressure loss in one of the tanks and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit, and maintain surveillance.
				Pressure switch fails closed	Launch to service module separation	Component rendered useless	One tank will over pressurize and be depleted before the other	Loss of redundant supply capability	Activate manual override circuit, and maintain surveillance.

Table 12. Cryogenic Storage Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Quantity and pressure control (Cont)				Resistor, open failure	Launch to service module separation	Component rendered useless	Pressure loss in one tank and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit, and maintain surveillance.
				Resistor, short failure	Launch to service module separation	Component rendered useless	One tank will over-pressure and be depleted before the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance.
				Resistor, out of tolerance	Launch to service module separation	Component usefullness marginal	This is unpredictable and the effects may range from sluggish operation of circuit to complete open condition	Variation of tank quantities out of 15 pound tolerance	Situation is marginal. Decision will have to be made on the basis of the seriousness of the problem at the time.
				Heater - open failure	Launch to service module separation	Component rendered useless	Pressure loss in one tank and subsequent depletion of the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance under abort conditions
				General signal control rectifier failure	Launch to service module separation	Component usefullness marginal	This is unpredictable and the effects may range from sluggish operation of circuit to complete open condition	Variation of tank quantities out of 15 pound tolerance	Situation is marginal. Decision will have to be made on the basis of the seriousness of the problem at the time.
				Warning light burn out	Launch to service module separation	Component rendered useless	None	None	Monitor pressure, temperature, and quantity gages.

Table 12. Cryogenic Storage Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Quantity and pressure control (Cont)				General transistor failure	Launch to service module separation	Component usefullness marginal	This is unpredictable and the effects may range from sluggish operation of the circuit to complete open condition	Variation of tank quantities out of 15 pound tolerance	Situation is marginal. Decision will have to be made on the basis of seriousness of the problem at the time.
				Short circuit (wiring)	Launch to service module separation	Component rendered useless	Loss of heater control or over pressurization and subsequent depletion of one tank before the other	Loss of redundant supply capability	Activate manual override circuit and maintain surveillance.
Squib valve	Beech 13532-4017		Close fuel cell supply line in the event of fuel cell failure	Inadvertent actuation	Launch to service module separation	None	One fuel cell will be shut down	None	None
Lines and fittings				Leakage	Launch to service module separation	Structurally weakened	Loss of gas	Insufficient gas to complete the 336 hour mission	Detect line or fitting and isolate it from rest of system by valve actuation if possible. Abort mission and maintain surveillance.
Tank	Beech 13532-4005		Store fluid in single phase condition	Leakage	Launch to service module separation	Structurally weakened	Loss of gas	Insufficient gas to complete the 336 hour mission	Detect tank and isolate it from rest of system by valve actuation if possible. Abort mission and maintain surveillance.

ELECTRICAL POWER SUBSYSTEM

During the period from October through December, 1962, effort has been directed toward a detailed systems analysis and the initiation of development tests on Apollo fuel cell hardware.

The present EPS reliability of 0.9941 closely approaches the system requirement of 0.9989.

A concentrated effort will be made regarding the fuel cell, primarily the electrode. Reliability of the electrode is assumed to be low because the item is newly developed and has no state-of-the-art history to support an accurate estimate. A major effort is being expended by the subcontractor, Pratt and Whitney, toward development of electrodes that meet the required reliability.

Results of all reliability analyses made to date, a summary of the electrode development program status, and a listing of projected activities planned for the next time period are included in this section.

CONFIGURATION ANALYSIS

Utilizing the EPS logic defined herein and the 14-day mission time, a preliminary reliability analysis has been completed. The results of this analysis delineate the requirements as well as the present estimate for EPS mission success reliability and show that the present inherent reliability is slightly less than the requirement. The major problem is the fuel cell electrode, which is summarized under Subcontractor Effort later in this section. Table 13 compares present reliability levels of EPS equipments (e.g., inverters, fuel cells, etc.) to equipments required to assure compatibility of the spacecraft, as well as EPS reliabilities with mission success requirements.

Table 13. Electrical Power System
Required Versus Estimated Reliability

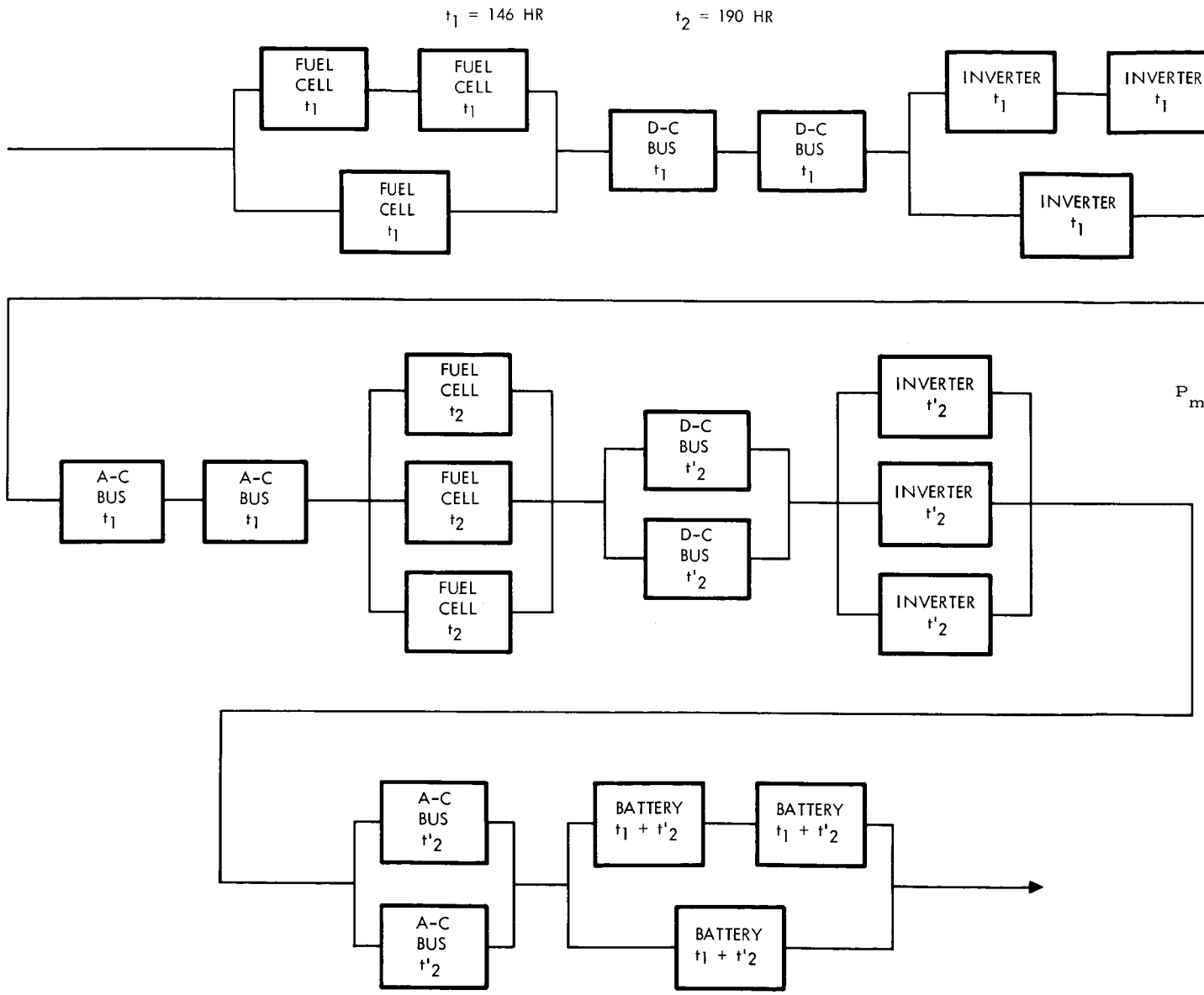
Subsystem	Reliability	
	Required	State-of-the-art
Fuel cell	0.971	0.9022
Inverter	0.9786	0.9638
Battery	0.9984	0.9984
A-c and d-c bus	0.99999	0.99999
Electrical power system	0.9989	0.9941

SUBSYSTEM DESCRIPTION

The Apollo electrical power system consists of four major subsystems: fuel cell, inverter, power distribution, and battery. (Reactant supply is supplied by the cryogenic storage subsystem; the reliability value as it affects the spacecraft is included in the section on the cryogenic storage subsystem.) Figure 15 presents the logic block diagram and mathematical model for the EPS and reflects the latest EPS abort and/or success criteria. A description of abort and/or success criteria for each subsystem is presented below. The operating times are: t_1 is the time to lunar orbit, and t_2 is the time from lunar orbit through service module separation or earth landing.

Fuel Cell

Primary d-c power is developed by a redundant fuel cell subsystem that functions from spacecraft lift-off to the initiation of the reentry phase (i.e., service module separation). This subsystem is composed of three individual cells, any two of which will provide sufficient power up to the lunar inject phase of the mission. After lunar orbit has been obtained, and for the remainder of the mission, only one cell out of three is required to ensure mission success and crew safety. Table 14 presents the reliability apportionment for fuel cell components.



$$P_{\text{mission success}} = \left\{ \frac{t_1}{P^3_{F/C}} \; \frac{t_2}{\left[P^3_{F/C} + 3P^2_{F/C}(1 - P_{F/C}) + 3P_{F/C}(1 - P_{F/C})^2\right]} \right.$$

$$\left. + \frac{t_1}{3P^2_{F/C}(1 - P_{F/C})} \; \frac{t_2}{\left[P^2_{F/C} + 2P_{F/C}(1 - P_{F/C})\right]} \right\} \left\{ \frac{t_1}{P^2_{\text{D-C BUS}}} \; \frac{t'_2}{\left[P^2_{\text{D-C BUS}} + 2P_{\text{D-C BUS}}(1 - P_{\text{D-C BUS}})\right]} \right\}$$

$$\left\{ \frac{t_1}{P^3_{INV}} \; \frac{t'_2}{\left[P^3_{INV} + 3P^2_{INV}(1 - P_{INV}) + 3P_{INV}(1 - P_{INV})^2\right]} + \frac{t_1}{3P^2_{INV}(1 - P_{INV})} \right.$$

$$\left. \frac{t'_2}{\left[P^2_{INV} + 2P_{INV}(1 - P_{INV})\right]} \right\} \left\{ \frac{t_1}{P^2_{\text{A-C BUS}}} \; \frac{t'_2}{\left[P^2_{\text{A-C BUS}} + P_{\text{A-C BUS}}(1 - P_{\text{A-C BUS}})\right]} \right\}$$

$$\left\{ \frac{t_1 + t'_2}{P^3_{BATT} + 3P^2_{BATT}(1 - P_{BATT})} \right\}$$

where
- t_1 = 146 hr
- t_2 = 190 hr
- $P_{F/C}$ = Fuel cell reliability
- P_{INV} = Inverter reliability
- P_{BATT} = Battery reliability
- $P_{\text{A-C BUS}}$ = A-c bus reliability
- $P_{\text{D-C BUS}}$ = D-c bus reliability

Figure 15. Electrical Power Subsystem Mission Success Logic Diagram

Table 14. Fuel Cell Component Reliability Apportionment

Component	Apportioned Reliability
Cell stack	0.9719
Motor, pump-separator, valve assembly	0.99975
Motor, pump assembly, glycol	0.999912
Nitrogen regulator	0.99986
Oxygen regulator	0.99988
Hydrogen regulator	0.99988
Regenerator bypass, hydrogen	0.999912
Accumulator	0.999976
Regenerator bypass, glycol	0.999956
Heat exchangers	0.9999966
Manifolds, tubing, and mechanical connections	0.9999978
Tank	0.9999989
Support and housing	0.9999989
Heater and wiring	0.99999984
Instrumentation	0.999999988
Purge valves	0.9999989
Relief valve	0.99999967
Fill valve	0.99999999981

Inverter

The d-c power furnished by the fuel cells is converted to ac in the inverter subsystem. This subsystem consists of three inverters; any one of these inverters will furnish sufficient a-c power to meet mission requirements. Stand-by redundancy is employed in the inverter subsystem, thus resulting in a more simplified design with higher reliability. If inverter failure occurs, a-c power can be restored by manually switching a stand-by inverter on the line.

Power Distribution

Power distribution, both ac and dc, is accomplished by individual, redundant bus subsystems. Both subsystems are essentially of the same design; they consist of two busses, either one being capable of distributing power to all spacecraft systems. Should failure occur on either bus or on connected systems, isolation may be effected and power delivered by the redundant bus. The design of these two distribution systems ensures high inherent reliability.

Battery

During spacecraft flight up to initiation of the reentry phase, secondary or emergency power is available from a battery subsystem. If the primary power subsystem fails, or if excessive power is required, the secondary system is automatically available and will meet load requirements for a limited time. However, the use of this subsystem as a secondary power source is not its prime function. This subsystem is intended as a primary power source during the reentry and post landing phases of the mission. It consists of three batteries in parallel; any two will meet the power requirements during reentry and up to the time of spacecraft recovery. With the use of redundancy and simplified design, the reliability of this subsystem is assured with a high degree of confidence.

RELIABILITY CALCULATION

Figure 15 presents the mathematical equation for EPS mission success. Crew safety reliabilities for the case of one of three fuel cells and one of three inverters successfully operating and one of two busses successfully operating are determined by evaluation of the math models shown in the Abort Criteria section.

[redacted]

FAILURE EFFECTS ANALYSIS

Table 15 at the end of this section presents the failure effects analysis of the power distribution system as described previously. It is apparent from this analysis that should a first order failure occur, electrical power will still be available to all spacecraft systems. Additional effort will be directed toward up-dating and expanding this analysis during the next report period. All changes, both additions and corrections, will appear on subsequent revisions of this analysis, which will be issued quarterly.

TEST PROGRAM

Fuel Cells

Pratt & Whitney submitted a qualification-reliability test plan on 15 November 1962; this document is currently being reviewed. The general philosophy is acceptable although revisions and additions to the mission life tests need to be made to provide a closer simulation of the mission.

Design and development testing of all components has begun. Figure delineates total development testing on the fuel cell. Development testing of the first independent fuel cell was completed in October. Problems were encountered with the potasium hydroxide; however, the cells were not sealed as they will be in service. The potasium hydroxide was contained in cups with tubes leading into each cell. Resulting potasium hydroxide spillage caused a short and corrosion. The voltage produced by each cell during this test was slightly in excess of the 1-volt design requirement. Development testing is scheduled for completion in June 1964.

Qualification testing of the fuel cell powerplant is scheduled to begin in June 1964. This date presents a 6-month delay from the date specified in the July-September report.

Effects of Qualification Program Re-orientation

The October redirection defined the number of cells for qualification testing as four in contrast to the original test plan prepared by Pratt & Whitney, which specified fifteen cells. The following changes in the tests were also made during the redirection. Mission simulation tests were expanded to include all mission environments, including ground checkout. All climatic tests except humidity were eliminated, and off-limit tests, including vibration, operational temperature, acoustics, and acceleration were added.

SUBCONTRACTOR MANAGEMENT

During the past quarter, S&ID Reliability Engineering reviewed type I documentation, which was submitted by Pratt & Whitney. Major milestones accomplished by Pratt & Whitney during this quarter were the initiation of powerplant and component test plans. During this quarter, all type I documentation was resubmitted and is currently being reviewed by S&ID Reliability Engineering.

A field analysis of the Pratt & Whitney cost proposal was conducted during the week of 26 November 1962. The purpose of the field analysis was to attain detailed information of all costs that are to be incurred for reliability on the fuel cell program.

A technical coordination meeting was held with Pratt & Whitney and representatives of S&ID during November for purposes of clarifying requirements of the procurement specification.

Subcontractor Effort

A summary of major accomplishments during this reporting period is given in the following paragraphs.

Electrodes have been tested in unpressurized rigs prior to assembly and use in multicell tests. Analysis of the results from these engineering tests have promulgated recommended tests and changes in inspection procedures in order to determine the reason for variations in bubble-pressure characteristics of Clevite-supplied electrodes. It was learned that some hydrogen sinter bubble test failures were caused by faulty resistance frazing rather than faulty sinters; corrective action is being taken.

A comprehensive program is being planned to coordinate all single cell inspection and performance data, to isolate the effects of measurable physical properties and process variables on performance, to assess the usefulness of destructive testing on used electrodes and waste corners (resulting from the fabrication of round electrodes from square stock), and to determine the effects of air and normal contaminants such as carbon dioxide on cell properties.

Details of the test plan for endurance and accelerated life tests on bearings is being prepared, using MIL-R-26667A as a guide. The outline will be used to set up a test plan that will investigate bearing life versus a simulated Apollo environment. The causes of failure and the mean-time-between-failure will be determined.

A number of designs were reviewed for reliability, including the heater power cable, heater harness installation, tank, reactant gas tubes, hydrogen pump-separator, primary regenerator, and tubing hardware. Several design modifications were incorporated as a result of these reviews to improve reliability. A number of quality assurance data sheets were checked and corrected where necessary.

A Monte Carlo analysis was performed on the IBM 7090 for the PC3A-2 system for three levels of reliability: 0.9785, which experienced ten single module failures in 200 missions; 0.898 with 16 single module failures and 2 double failures in 50 missions; and 0.880 with 17 single and three double module failures in 50 missions. All failures except two occurred in cells. In no cases did all three modules fail during the 400-hour missions. Details will be given in the next Quarterly Reliability Status Report.

A list of parts that do not require traceability is being prepared, as directed by an S&ID letter M3MA1-6-823 dated 5 November 1962 on the interpretation of NCP 200-2, sections 4.4 and 5.7.

PLANNED ACTIVITIES

During the next reporting period, effort will be placed on up-dating the electrical system failure mode analysis, along with component reliability values. Because contract awards were made on the inverter, battery, and battery charger during this reporting period, adequate detail was not available for inclusion in this report. During the next reporting period, detailed failure mode analysis, apportionments, and state-of-the-art values will be investigated, along with improvements of system design on the aforementioned units.

Development tests on the fuel cell will have progressed to a point where initial assessments will be made on fuel cell components.

Table 15. Electrical Power System Failure Effects Analysis

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Relay	V14-945101 dated 10/25/62	K5	Inverter 1 to bus 1 relay	Fails open		Relay inoperative	No ac power available at bus 1 while inverter 1 is on	None	Loads can be switched to bus 2, or inverter 2 or 3 can be switched in.
				Fails short		Relay inoperative	No effect	None	In the event of a short on bus 1, circuit breakers CB65, CB67 and CB69 can be opened and power can be switched to 2.
Relay	V14-945101 dated 10/25/62	K4	Inverter 1 to bus 2 relay	Fails open		Relay inoperative	No ac power available at bus 1 while inverter 1 is on	None	Loads can be switched to bus 1, or inverter 2 or 3 can be switched in.
				Fails short		Relay inoperative	No effect	None	In the event of a short on bus 2, circuit breakers CB64, CB66 and CB68 can be opened and power can be switched to bus 1.
Circuit breaker	V14-945101 dated 10/25/62	CB65	Inverter 1, $\emptyset A_1$, output to bus 1 circuit breaker (short circuit protection for inverter 1, $\emptyset A_1$)	Fails open		Loss of circuit breaker	No ac power available at $\emptyset$A, bus 1, while inverter 1 is on	None	Loads can be switched to bus 2, or inverter 2 or 3 switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 1, A1, from $\emptyset$A, bus 1	None	In the event of a short on $\emptyset$A, bus 1, inverter 1 can be shut off and inverter 2 or 3 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB64	Inverter 1, $\emptyset A_2$, output to $\emptyset A$ bus 2 circuit breaker (short circuit protection for inverter 1 $\emptyset A_2$)	Fails open		Loss of circuit breaker	No ac power available at $\emptyset A$, bus 2, while inverter 1 is on.	None	Loads can be switched to bus 1, or inverter 2 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 1, $\emptyset A2$, from $\emptyset A$, bus 2	None	In the event of $\emptyset A$ short on a bus 2, inverter 1 can be shut off and inverter 2 or 3 can be switched in.
Relay	V14-945101 dated 10/25/62	K1	DC input power relay to inverter 1	Fails open		Relay inoperative	Inverter 1 would not be available.	None	Inverter 2 or 3 can be switched in to restore power.
				Fails short		Relay inoperative	No effect	None	In the event of a short, inverter 1 can be shut off and inverter 2 or 3 can be switched in.
Relay	V14-945101 dated 10/25/62	K2	DC input power relay to inverter 2	Fails open		Relay inoperative	Inverter 2 would not be available.	None	Inverter 1 or 3 can be switched in to restore power.
				Fails short		Relay inoperative	No effect	None	In the event of a short, inverter 2 can be shut off and inverter 1 or 3 can be switched in.
Relay	V14-945101 dated 10/25/62	K3	DC input power relay to inverter 3	Fails open		Relay inoperative	Inverter 3 would not be available.	None	Inverter 1 or 2 can be switched in to restore power.
				Fails short		Relay inoperative	No effect	None	In the event of $\emptyset A$ short, inverter 3 can be shut off and inverter 1 or 2 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB71	Inverter 2, $\emptyset A_1$, output to $\emptyset A$, bus 1 circuit breaker (short circuit protection for inverter 2 $\emptyset A_1$)	Fails open		Loss of circuit breaker	No ac power available at $\emptyset A$, bus 1, while inverter 2 is on	None	Loads can be switched to bus 2, or inverter 1 or 3 can be switched in.
				Fails short		Loss of circuit breakers	No short circuit protection for inverter 2, $\emptyset A_1$, from $\emptyset A$, bus 1	None	In the event of a short on bus 1, inverter 2 can be shut off and inverter 1 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB70	Inverter 2, $\emptyset A_2$, output to $\emptyset A$, bus 2 circuit breaker (short circuit protection for inverter 2 $\emptyset A_2$)	Fails open		Loss of circuit breaker	No ac power available at $\emptyset A$, bus 2, while inverter 2 is on	None	Loads can be switched to bus 1, or inverter 1 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 2, $\emptyset A_2$, from $\emptyset A$, bus 2	None	In the event of a short on $\emptyset A$, bus 2, inverter 2 can be shut off and inverter 1 or 3 can be switched in.
						Loss of circuit breaker	No ac power available at $\emptyset A$, bus 1, while inverter 3 is on	None	Loads can be switched to bus 2, or inverter 1 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB77	Inverter 3, $\emptyset A_1$, output to $\emptyset A$, bus 1 circuit breaker (short circuit protection for inverter 3 $\emptyset A_1$)	Fails open		Loss of circuit breaker	No short circuit protection for inverter 3, $\emptyset A_1$, from $\emptyset A$, bus 1	None	In the event of a short on $\emptyset A$, bus 1, inverter 3 can be shut off and inverter 1 or 2 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 3, $\emptyset A_1$, from $\emptyset A$, bus 1	None	In the event of a short on $\emptyset A$, bus 1, inverter 3 can be shut off and inverter 1 or 2 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB76	Inverter 3, $\emptyset A_2$, output to $\emptyset A$, bus 2 circuit breaker (short circuit protection for inverter 3 $\emptyset A_2$)	Fails open		Loss of circuit breaker	No ac power available at $\emptyset A$, bus 2, while inverter 3 is on	None	Loads can be switched to bus 1, or inverter 1 or 2 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 3 from $\emptyset A$, bus 2	None	In the event of a short on $\emptyset A$, bus 2, inverter 3 can be shut off and inverter 1 or 2 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB67	Inverter 1, $\emptyset B_1$, output to $\emptyset B$, bus 1 circuit breaker (short circuit protection for inverter 1 $\emptyset B_1$)	Fails open		Loss of circuit breaker	No ac power available to $\emptyset B$, bus 1, while inverter 1 is on	None	Loads can be switched to bus 2, or inverter 2 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 1, $\emptyset B_1$, from $\emptyset B$, bus 1	None	In the event of a short on $\emptyset B$, bus 1, inverter 1 can be shut off and inverter 2 or 3 can be switched in
Circuit breaker	V14-945101 dated 10/25/62	CB66	Inverter 1, $\emptyset B_2$, output to $\emptyset B$, bus 2 circuit breaker (short circuit protection for inverter 1 $\emptyset B_2$)	Fails open		Loss of circuit breaker	No ac power available to $\emptyset B$, bus 2, while inverter 1 is on	None	Loads can be switched to bus 1, or inverter 2 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 1, $\emptyset B_2$, from $\emptyset B$, bus 2	None	In the event of a short on $\emptyset B$, bus 2, inverter 1 can be shut off and inverter 2 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB73	Inverter 2, $\emptyset B_1$, output to $\emptyset B$, bus 1 circuit breaker (short circuit protection for inverter 2, $\emptyset B_1$)	Fail open		Loss of circuit breaker	No ac power available to $\emptyset B$, bus 1, while inverter 2 is on	None	Loads can be switched to bus 2, or inverter 1 or 3 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker (Cont)	V14-945101 dated 10/25/62	CB73		Fail short		Loss of circuit breaker	No short circuit protection for inverter 2, $\emptyset B_1$, from $\emptyset B$, bus 1	None	In the event of a short on $\emptyset B$, bus 1, inverter 2 can be shut off and inverter 1 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB72	Inverter 2, $\emptyset B_2$, output to $\emptyset B$, bus 2 circuit breaker (short circuit protection for inverter 2 $\emptyset B_2$)	Fails open		Loss of circuit breaker	No ac power available to $\emptyset B$, bus 2, while inverter 2 is on	None	Loads can be switched to bus 1, or inverter 1 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 2 from $\emptyset B$, bus 2	None	In the event of a short on $\emptyset B$, bus 2, inverter 2 can be shut off and inverter 1 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB79	Inverter 3, $\emptyset B_1$, output to $\emptyset B$, bus 1 circuit breaker (short circuit protection for inverter 3, $\emptyset B_1$)	Fails open		Loss of circuit breaker	No ac power available to $\emptyset B$, bus 1, while inverter 3 is on	None	Loads can be switched to bus 2, or inverter 1 or 2 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 3, $\emptyset B_1$, from $\emptyset B$, bus 1	None	In the event of a short on $\emptyset B$, bus 1, inverter 3 can be shut off and inverter 1 or 2 can be switched on.
Circuit breaker	V14-945101 dated 10/25/62	CB78	Inverter 3, $\emptyset B_2$, output to $\emptyset B$, bus 2 circuit breaker (short circuit protection for inverter 3, $\emptyset B_2$)	Fails open		Loss of circuit breaker	No ac power available to $\emptyset B$, bus 2, while inverter 3 is on	None	Loads can be switched to bus 1, or inverter 1 or 2 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 3, $\emptyset B_2$, from $\emptyset B$, bus 2	None	In the event of a short on $\emptyset B$, bus 2, inverter 3 can be shut off and inverter 1 or 2 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB81	Inverter 3, ØC1, output to ØC, bus 1 circuit breaker (short circuit protection for inverter 3 $ØC_1$)	Fails open		Loss of circuit breaker	No ac power available to ØC, bus 1, while inverter 3 is on	None	Loads can be switched to bus 2, or inverter 1 or 2 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 3, $ØC_1$, from ØC, bus 1	None	In the event of a short on ØC, bus 1, inverter 3 can be shut off and inverter 1 or 2 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB80	Inverter 3, $ØC_2$, output to ØC, bus 2 circuit breaker (short circuit protection for inverter 3, $ØC_2$)	Fails open		Loss of circuit breaker	No ac power available to ØC, bus 2, while inverter 3 is on	None	Loads can be switched to bus 1, or inverter 1 or 2 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 3, ØC2, from ØC, bus 2	None	In the event of a short on ØC, bus 2, inverter 3 can be shut off and inverter 1 or 2 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB75	Inverter 2, $ØC_1$, output to ØC, bus 1 circuit breaker (short circuit protection for inverter 3, ØC1)	Fails open		Loss of circuit breaker	No ac power available to ØC, bus 1, while inverter 2 is on	None	Loads can be switched to bus 2, or inverter 1 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 2, $ØC_1$, from ØC, bus 1	None	In the event of a short on ØC, bus 1, inverter 2 can be shut off and inverter 1 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB74	Inverter 2, $ØC_2$, output to ØC, bus 2 circuit breaker (short circuit protection for inverter 2, $ØC_2$)	Fails open		Loss of circuit breaker	No ac power available to ØC, bus 2, while inverter 2 is on	None	Loads can be switched to bus 1, or inverter 1 or 3 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker (Cont)	V14-945101 dated 10/25/62	CB74		Fails short		Loss of circuit breaker	No short circuit protection for inverter 2, ØC2, from ØC, bus 2	None	In the event of a short on ØC, bus 2, inverter 2 can be shut off and inverter 1 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB69	Inverter 1, $ØC_1$, output to ØC, bus 1 circuit breaker (short circuit protection for inverter 1, $ØC_1$)	Fails open		Loss of circuit breaker	No ac power available to ØC, bus 1, while inverter 1 is on.	None	Loads can be switched to bus 2, or inverter 2 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 1, $ØC_1$, from ØC, bus 1	None	In the event of a short on ØC, bus 1, inverter 1 can be shut off and inverter 2 or 3 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB68	Inverter 1, $ØC_2$, output to ØC, bus 2 circuit breaker (short circuit protection for inverter 1, $ØC_2$)	Fails open		Loss of circuit breaker	No ac power available to ØC, bus 2, while inverter 1 is on	None	Loads can be switched to bus 1, or inverter 2 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 1, $ØC_2$, from ØC, bus 2	None	In the event of a short on ØC, bus 2, inverter 1 can be shut off and inverter 2 or 3 can be switched in.
Relay	V14-945101 dated 10/25/62	K7	Inverter 2 to bus 1 relay	Fail open		Loss of relay	No ac power available at bus 1 while inverter 2 is on	None	Loads can be switched to bus 2, or inverter 1 or 3 can be switched in.
				Fail short		Loss of relay	No effect	None	In the event of a short on bus 1, circuit breakers CB71, CB73, and CB75 can be opened and power can be switched to bus 2.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Relay	V14-945101 dated 10/25/62	K9	Inverter 3 to bus 1 relay	Fail open		Loss of relay	No ac power available at bus 1 while inverter 3 is on	None	Loads can be switched to bus 2, or inverter 1 or 2 can be switched in.
				Fail short		Loss of relay	No effect	None	In the event of a short on bus 1, circuit breakers CB81, CB79, CB77 can be opened and power can be switched to bus 2.
Relay	V14-945101 dated 10/25/62	K6	Inverter 2 to bus 2 relay	Fail open		Loss of relay	No ac power available at bus 2 while inverter 2 is on	None	Loads can be switched to bus 1, or inverter 1 or 3 can be switched in.
				Fail short		Loss of relay	No effect	None	In the event of a short on bus 2, circuit breakers CB72, CB70, and CB74 can be opened and power can be switched to bus 1.
Relay	V14-945101 dated 10/25/62	K8	Inverter 3 to bus 2 relay	Fail open		Loss of relay	No ac power available at bus 2 while inverter 3 is on	None	Loads can be switched to bus 1, or inverter 1 or 2 can be switched in.
				Fail short		Loss of relay	No effect	None	In the event of a short on bus 2, circuit breakers CB80, CB78, and CB76 can be switched to bus 1.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
AC bus ØA bus 1	V14-945101 dated 10/25/62		Distributes power	Fail open		Loss of bus	No ac power available at ØA, bus 1	None	Loads can be switched to bus 2.
				Fail short		Loss of bus	No ac power available at ØA, bus 1	None	Loads can be switched to bus 2.
AC bus ØB bus 1	V14-945101 dated 10/25/62		Distributes power	Fail open		Loss of bus	No ac power available at ØB, bus 1	None	Loads can be switched to bus 2.
				Fail short		Loss of bus	No ac power available at ØB, bus 1	None	Loads can be switched to bus 2.
AC bus ØC bus 1	V14-945101 dated 10/25/62		Distributes power	Fail open		Loss of bus	No ac power available at ØC, bus 1	None	Loads can be switched to bus 2.
				Fail short		Loss of bus	No ac power available at ØC, bus 1	None	Loads can be switched to bus 2.
AC bus ØA bus 2	V14-945101 dated 10/25/62		Distributes power	Fail open		Loss of bus	No ac power available at ØA, bus 2	None	Loads can be switched to bus 1.
				Fail short		Loss of bus	No ac power available at ØA, bus 2	None	Loads can be switched to bus 1.
AC bus ØB bus 2	V14-945101 dated 10/25/62		Distributes power	Fail open		Loss of bus	No ac power available at ØB, bus 2	None	Loads can be switched to bus 1.
				Fail short		Loss of bus	No ac power available at ØB, bus 2	None	Loads can be switched to bus 1.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
AC bus ØC bus 2	V14-945101 dated 10/25/62		Distributes power	Fail open		Loss of bus	No ac power available at ØC, bus 2	None	Loads can be switched to bus 1.
				Fail short		Loss of bus	No ac power available at ØC, bus 2	None	Loads can be switched to bus 1.
Circuit breaker	V14-945101 dated 10/25/62	CB83	AC bus to fuel cell 1 circuit breaker	Fail open		Loss of circuit breaker	No ac power available to fuel cell 1.	None	None
				Fail short		Loss of circuit breaker	No short circuit protection for fuel cell 1 motors	None	None
Circuit breaker	V14-945101 dated 10/25/62	CB84	AC bus to fuel cell 2 circuit breaker	Fail open		Loss of circuit breaker	No ac power available to fuel cell 2	None	None
				Fail short		Loss of circuit breaker	No short circuit protection for fuel cell 2 motors	None	None
Circuit breaker	V14-945101 dated 10/25/62	CB85	AC bus to fuel cell 3 circuit breaker	Fail open		Loss of circuit breaker	No ac power available to fuel cell 3	None	None
				Fail short		Loss of circuit breaker	No short circuit protection for fuel cell 3 motors	None	None
Inverter 1	V14-945101 dated 10/25/62		Supplies AC power	No power output	Flight	Loss of inv	No ac power available to bus A or B	None	Switch standby inverter on
Inverter 2	V14-945101 dated 10/25/62		Supplies AC power	No power output	Flight	Loss of inv	No ac power available to bus A or B	None	Switch standby inverter on

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Inverter 3	V14-945101 dated 10/25/62		Supplies AC power	No power output	Flight	Loss of inv	No ac power available to bus A or B	None	Switch standby inverter on
Circuit breaker	V14-945101 dated 10/25/62	CB47	Post landing battery C to post landing bus circuit breaker	Fails open		Loss of circuit breaker	No dc power to post landing bus from battery C	None	Battery A or B can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for battery C	None	Battery A or B can be switched in if battery C is shorted out.
Post landing Battery C	V14-945101 dated 10/25/62		Supplies DC power to post landing bus	No power output		Loss of battery C	No dc power available to post landing bus	None	Battery A or B can be switched in.
Battery A	V14-945101 dated 10/25/62		Supplies dc power to battery bus A	No power output		Loss of battery A	No dc power available to battery bus A	None	Battery B or C can be switched in.
Battery B	V14-945101 dated 10/25/62		Supplies dc power to battery bus B	No power output		Loss of battery B	No dc power available to battery bus B	None	Battery A or C can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB45	Battery A to battery bus A circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to battery bus A	None	Battery B or C can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery A	None	Battery B or C can be switched in if battery A is shorted out.
Circuit breaker	V14-945101 dated 10/25/62	CB46	Battery B to bus B circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to battery bus B	None	Battery A or C can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery B	None	Battery A or C can be switched in if battery B is shorted out.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Battery bus A	V14-945101 dated 10/25/62		Distributes power	Fails open		Loss of bus	No dc power at battery bus A	None	Power can be switched to battery bus B or post landing bus.
				Fails short		Loss of bus	No dc power at battery bus A	None	Power can be switched to battery bus B or post landing bus.
Battery bus B	V14-945101 dated 10/25/62		Distributes power	Fails open		Loss of bus	No dc power at battery bus B	None	Power can be switched to battery bus A or post landing bus.
				Fails short		Loss of bus	No dc power at battery bus B	None	Power can be switched to battery bus A or post landing bus.
Circuit breaker	V14-945101 dated 10/25/62	CBB	DC main bus A to inverter 1 circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to inverter 1	None	Inverter 2 or 3 can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for inverter 1 input from dc main bus A	None	Inverter 2 or 3 can be switched in if inverter 1 is shorted out.
Diode	V14-945101 dated 10/25/62	CR17	Provides path for current flow dc main bus A to battery charger; also isolates dc main bus A from dc main bus B	Fails open		Diode inoperative	No dc power available to battery charger from dc main bus A	None	DC main bus B can be switched in to provide input power to battery charger.
				Fails short		Diode inoperative	No isolation between dc main bus A and dc main bus B	None	Circuit breaker CB 5 can be opened.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Diode	V14-945101 dated 10/25/62	CR18	Provides path for current flow from dc main bus B to battery charger; also isolates dc main bus B from dc main bus A	Fails open		Diode inoperative	No dc power available to battery charger from dc main bus B	None	DC main bus A can be switched in to provide input power to battery charger.
				Fails short		Diode inoperative	No isolation between dc main bus B and dc main bus A	None	Circuit breaker CB6 can be opened.
Circuit breaker	V14-945101 dated 10/25/62	CB5	DC main bus A to battery charger circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to battery charger from dc main bus A	None	Power can be switched to dc main bus B
				Fails short		Circuit breaker inoperative	No short circuit protection to battery charger from dc main bus A	None	Power can be switched to dc main bus B if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB6	DC main bus B to battery charger circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to battery charger from dc main bus B	None	Power can be switched to dc main bus A.
				Fails short		Circuit breaker inoperative	No short circuit protection to battery charger from dc main bus B	None	Power can be switched to dc main bus B if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB34	Battery bus B to dc main bus A circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to dc main bus A from battery bus B	None	DC main bus B can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus B from dc main bus A	None	DC main bus B can be switched in if a short occurs.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Relay	V14-945101 dated 10/25/62	K16	Battery bus B to dc main bus A relay	Fails open		Relay inoperative	No dc power available to dc main bus A or dc main bus B from battery bus B	None	Battery bus A can be switched in.
				Fails short		Relay inoperative	No short circuit protection for battery bus B from dc main bus A	None	Battery bus A can be switched in if a short occurs.
Diode	V14-945101 dated 10/25/62	CR9	Provides path for current flow from battery bus B to dc main bus A; also isolates battery bus B from dc main bus A	Fails open		Diode inoperative	No dc power available to dc main bus A from battery bus B	None	Battery bus A can be switched in or dc main bus B can be switched in.
				Fails short		Diode inoperative	No isolation between battery bus B and dc main bus A	None	Circuit breaker CE34 can be opened.
Circuit breaker	V14-945101 dated 10/25/62	CB36	Battery bus B to dc main bus B circuit breaker	Fails open		Circuit breaker inoperative	No dc power available from battery bus B to dc main bus B	None	DC main bus A can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus B from dc main bus B	None	DC main bus A can be switched in if a short occurs.
Diode	V14-945101 dated 10/25/62	CR10	Provides path for current flow from battery bus B to dc main bus B; also isolates battery bus B from dc main bus B	Fails open		Diode inoperative	No dc power available to dc main bus B from battery bus B	None	Battery bus A can be switched in or dc main bus A can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB35	Battery bus A to dc main bus B circuit breaker	Fails open		Circuit breaker inoperative	No dc power available from battery bus A to dc main bus B	None	DC main bus A can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus A from dc main bus B	None	DC main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB37	Battery bus A to dc main bus A circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to dc main bus A from battery bus A	None	DC main bus B can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus A from dc main bus A	None	DC main bus B can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB37	Battery bus A to pyro bus A circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to pyro bus A from battery bus A	None	Battery bus B can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus A from pyro bus A	None	Battery bus B and pyro bus B can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB43	Battery bus A to post landing bus circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to post landing bus from battery bus A	None	Battery bus B can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus A from pyro bus A	None	Battery bus B can be switched in if a short occurs.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB 39	Battery bus A to battery relay bus circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to battery relay bus from battery bus A	None	Battery bus B can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus A from battery relay bus	None	Battery bus B can be switched in if a short occurs.
Diode	V14-945101 dated 10/25/62	CR15	Provides path for current flow from battery bus A to battery relay bus; also isolates battery bus A from battery relay bus	Fails open		Diode inoperative	No dc power available from battery bus A to battery relay bus	None	Battery bus B can be switched in.
				Fails short		Diode inoperative	No isolation between bus A and battery relay bus	None	Battery bus B can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB38	Battery bus B to pyro bus B circuit breaker	Fails open		Circuit breaker inoperative	No dc power available from battery bus B to pyro bus B	None	Battery bus A and pyro bus A can be switched in if a short occurs.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus B from pyro bus B	None	Battery bus A and pyro bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB44	Battery bus B to post landing bus circuit breaker	Fails open		Circuit breaker inoperative	No dc power available in post landing bus from battery bus B	None	Battery bus A can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus B from post landing bus	None	Battery bus A can be switched in if a short occurs.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB40	Battery bus B to battery relay bus circuit breaker	Fails open		Circuit breaker inoperative	No dc power available to battery relay bus from battery bus B	None	Battery bus A can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for battery bus B from battery relay bus	None	Battery bus A can be switched in if a short occurs.
Diode	V14-945101 dated 10/25/62	CR16	Provides path for current flow from battery bus B to battery relay bus; also isolates battery bus B from battery relay bus	Fails open		Diode inoperative	No dc power available from battery bus B to battery relay bus	None	Battery bus A can be switched in.
				Fails short		Diode inoperative	No isolation between battery bus B and battery relay bus	None	Battery bus A can be switched in.
Diode	V14-945101 dated 10/25/62	CR7	Provides path for current flow from battery bus A to dc main bus A; also isolates battery bus A from DC main bus A	Fails open		Diode inoperative	No dc power available to dc main bus A from battery bus A	None	Battery bus B can be switched in.
				Fails short		Diode inoperative	No isolation between battery bus A and dc main bus A	None	Battery bus B can be switched in.
Diode	V14-945101 dated 10/25/62	CR8	Provides path for current flow from battery bus A to dc main bus B; also isolates battery bus A from dc main bus B	Fails open		Diode inoperative	No dc power available to dc main bus B from battery bus A	None	Battery bus B can be switched in.
				Fails short		Diode inoperative	No isolation between battery bus A and dc main bus B	None	Battery bus B can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB3	DC main bus A to post landing bus, circuit breaker	Fails open		Relay inoperative	No dc power available to post landing bus from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Relay inoperative	No short circuit protection for dc main bus A from post landing bus	None	DC main bus B can be switched in if a short occurs.
Diode	V14-945101 dated 10/25/62	CR13	Provides path for current flow from dc main bus A to post landing bus; also isolates post landing bus from dc main bus A	Fails open		Diode inoperative	No dc power available to post landing bus from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Diode inoperative	No isolation between post landing bus and dc main bus A	None	DC main bus B can be switched in.
DC main bus B	V14-945101 dated 10/25/62		Distributes power	Fails open		Loss of bus	No dc power available at dc main bus B	None	DC main bus A can be switched in.
				Fails short		Loss of bus	No dc power available at dc main bus B	None	DC main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB15	DC main bus A to inverter 3 circuit breaker	Fails open		Loss of circuit breaker	No dc power available to inverter 3	None	Inverter 1 or 2 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for inverter 3 from dc main bus A	None	Inverter 1 or 2 can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB14	DC main bus B to inverter 2 circuit breaker	Fails open		Loss of circuit breaker	No dc power available to inverter 2	None	Inverter 1 or 3 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker				Fails short		Loss of circuit breaker	No short circuit protection for inverter 2 from dc main bus B	None	Inverter 1 or 3 can be switched in if a short occurs.
Switch	V14-945101 dated 10/25/62	S19	ON-OFF switch for inverter 3 dc power input	Fails open		Loss of switch	No dc power available to inverter 3	None	Inverter 1 or 2 can be switched in.
				Fails closed position 1		Loss of switch	No dc power available to inverter 3 from dc main bus B	None	DC main bus A can be switched in, or inverter 1 or 2 can be switched in.
				Fails closed position 2		Loss of switch	No dc power available to inverter 3 from dc main bus A	None	DC main bus B can be switched in, or inverter 1 or 2 can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB4	DC main bus B to post landing bus circuit breaker	Fails open		Circuit breaker inoperative	No power available to post landing bus from dc main bus B	None	DC main bus A can be switched in.
				Fails short		Circuit breaker inoperative	No short circuit protection for dc main bus B from post landing bus	None	DC main bus A can be switched in if a short occurs.
Diode	V14-945101 dated 10/25/62	CR14	Provides path for current flow from dc main bus B to post landing bus; also isolates post landing bus from dc line bus B	Fails open		Diode inoperative	No power available from dc main bus B to post landing bus	None	DC main bus A can be switched in.
				Fails short		Diode inoperative	No isolation between post landing bus and dc main bus B	None	DC main bus A can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Pyro bus A	V14-945101 dated 10/25/62		Distributes power	Fails open		Loss of bus	No dc power available at pyro bus A	None	Pyro bus B can be switched in.
				Fails short		Loss of bus	No dc power available at pyro bus A	None	Pyro bus B can be switched in if a short occurs.
Pyro bus B	V14-945101 dated 10/25/62		Distributes power	Fails open		Loss of bus	No dc power available at pyro bus B	None	Pyro bus A can be switched in.
				Fails short		Loss of bus	No dc power available at pyro bus B	None	Pyro bus A can be switched in if a short occurs.
DC main bus A	V14-945101 dated 10/25/62		Distributes power	Fails open		Loss of bus	No dc power available at dc main bus A	None	DC main bus B can be switched in.
				Fails short		Loss of bus	No dc power available at dc main bus A	None	DC main bus B can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB17	DC main bus A to battery relay bus circuit breaker	Fails open		Loss of circuit breaker	No dc power available to battery relay bus from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus A from battery relay bus	None	DC main bus B can be switched if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB18	DC main bus B to battery relay bus circuit breaker	Fails open		Loss of circuit breaker	No dc power available to battery relay bus from dc main bus B	None	DC main bus A can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker (Cont)				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus B battery relay bus	None	DC main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB23	DC main bus A to navigation and guidance circuit breaker	Fails open		Loss of circuit breaker	No dc power available to navigation and guidance from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus A from navigation and guidance	None	DC main bus B can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB24	DC main bus B to navigation and guidance circuit breaker	Fails open		Loss of circuit	No dc power available to navigation and guidance from dc main bus B	None	DC main bus A can be switched in.
				Fails short		Loss of circuit	No short circuit protection for dc main bus B from navigation and guidance	None	DC main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB25	DC main bus A to telecommunications circuit breaker	Fails open		Loss of circuit breaker	No dc power available to telecommunication from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus A from telecommunication	None	DC main bus B can be switched in if a short occurs.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker	V14-945101 dated 10/25/62	CB26	DC main bus B to telecommunications circuit breaker	Fails open		Loss of circuit breaker	No dc power available to telecommunication from dc main bus B	None	DC main bus A can be switched in if a short occurs.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus B from telecommunication	None	DC main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB27	DC main bus A to stabilization and control circuit breaker	Fails open		Loss of circuit breaker	No dc power available to stabilization and control from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus A from stabilization and control	None	DC main bus B can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB28	DC main bus B to stabilization and control circuit breaker	Fails open		Loss of circuit breaker	No dc power available to stabilization and control from dc main bus B	None	DC main bus A can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus B from stabilization and control	None	DC main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB29	DC main bus A to reaction control circuit breaker	Fails open		Loss of circuit breaker	No dc power available to reaction control from dc main bus A	None	DC main bus B can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker (Cont)				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus A from reaction control	None	DC main bus B can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB30	DC main bus B to reaction control circuit breaker	Fails open		Loss of circuit breaker	No dc power available to reaction control from dc main bus B	None	DC main bus A can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus B from reaction control	None	DC main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB1	DC main bus A to main relay bus circuit breaker	Fails open		Loss of circuit breaker	No dc power available to main relay bus from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus A from relay bus	None	DC main bus B can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB2	DC main bus B to main relay bus circuit breaker	Fails open		Loss of circuit breaker	No dc power available to main relay bus from dc main bus B	None	DC main bus A can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for dc main bus B from main relay bus	None	DC main bus B can be switched in if a short occurs.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Diode	V14-945101 dated 10/25/62	CR11	Provides path for current flow from dc main bus A to main relay bus	Fails open		Loss of diode	No dc power available to main relay bus from dc main bus A	None	DC main bus B can be switched in.
				Fails short		Loss of diode	No isolation between dc main bus A and main relay bus	None	DC main bus B can be switched in.
Diode	V14-945101 dated 10/25/62	CR12	Provides path for current flow from dc main bus B to main relay bus	Fails open		Loss of diode	No dc power available to main relay bus from dc main bus B	None	DC main bus A can be switched in.
				Fails short		Loss of diode	No isolation between dc main bus B and main relay bus	None	DC main bus A can be switched in.
Circuit breaker	V14-945101 dated 10/25/62	CB61	Main relay bus to inverter 1 on-off switch circuit bre	Fails open		Loss of circuit breaker	No dc power available to inverter 1	None	Inverter 2 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for main relay bus from inverter 1	None	Inverter 2 or 3 can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB62	Main relay bus to inverter 2 on-off switch circuit breaker	Fails open		Loss of circuit breaker	No dc power available to inverter 2	None	Inverter 1 or 3 can be switched in.
				Fails short		Loss of circuit breaker	No short circuit protection for main relay bus from inverter 2	None	Inverter 1 or 3 can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB62	Main relay bus to inverter 3 on-off switch circuit breaker	Fails open		Loss of circuit breaker	No dc power available to inverter 3	None	Inverter 1 or 2 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker (Cont)				Fails short		Loss of circuit breaker	No short circuit protection for main relay bus from inverter 3	None	Inverter 1 or 2 can be switched in if a short occurs.
Switch	V14-945101 dated 10/25/62	S1	Inverter 1 on-off switch	Fails closed on position		Loss of switch	Inverter 1 cannot be turned off with switch S1.	None	Circuit breaker CB61 can be opened in event inverter 1 has to be shut off.
				Fails closed off position		Loss of switch	Inverter 1 cannot be turned on.	None	Inverter 2 or 3 can be switched in.
Switch	V14-945101 dated 10/25/62	S2	Inverter 2 on-off switch	Fails closed on position.		Loss of switch	Inverter 2 cannot be turned off with switch S2.	None	Circuit breaker CB62 can be opened in event inverter 2 has to be shut off.
				Fails closed off position		Loss of switch	Inverter 2 cannot be turned on.	None	Inverter 1 or 3 can be switched in.
Switch	V14-945101 dated 10/25/62	S3	Inverter 3 on-off switch	Fails closed on position		Loss of switch	Inverter 3 cannot be turned off with switch S3.	None	Circuit breaker CB63 can be opened in event inverter 3 has to be shut off.
				Fails closed off position		Loss of switch	Inverter 3 cannot be turned on.	None	Inverter 1 or 2 can be switched in.
Switch	V14-945101 dated 10/25/62	S4	Inverter 1 to bus 2 on-off switch	Fails closed on position		Loss of switch	With inverter 1 on ac, power cannot be removed from bus 2 by switch S4.	None	In the event of a short circuit, inverter 2 or 3 can be switched in.
				Fails closed off position		Loss of switch	No ac power available to bus 2 while inverter 1 is on	None	Inverter 2 or 3 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Switch	V14-945101 dated 10/25/62	S5	Inverter 1 to bus 1 on-off switch	Fails closed on position		Loss of switch	With inverter 1 on AC, power cannot be removed from bus 1 by switch S5.	None	In the event of a short circuit, inverter 2 or 3 can be switched in.
				Fails closed off position		Loss of switch	No ac power available to bus 1 while inverter 1 is on.	None	Inverter 2 or 3 can be switched in.
Switch	V14-945101 dated 10/25/62	S6	Inverter 2 to bus 2 on-off switch	Fails closed on position		Loss of switch	With inverter 2 on ac, power cannot be removed from bus 2 by switch S6.	None	In the event of a short circuit, inverter 1 or 3 can be switched in.
				Fails closed off position		Loss of switch	No ac power available to bus 2 while inverter 2 is on.	None	Inverter 1 or 3 can be switched in.
Switch	V14-945101 dated 10/25/62	S7	Inverter 2 to bus 1 on-off switch	Fails closed on position		Loss of switch	With inverter 2 on ac, power cannot be removed from bus 1 by switch S7.	None	In the event of a short circuit, inverter 1 or 3 can be switched in.
Switch	V14-945101 dated 10/25/62	S8	Inverter 3 to bus 2 on-off switch	Fails closed off position		Loss of switch	No ac power available to bus 1 while inverter 2 is on.	None	Inverter 1 or 3 can be switched in.
				Fails closed on position		Loss of switch	With inverter 3 on, ac power cannot be removed from bus 2 by switch S8.	None	In the event of a short circuit, inverter 1 or 2 can be switched in.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Switch (Cont)				Fails closed off position		Loss of switch	No ac power available to bus 2 while inverter 3 is on.	None	Inverter 1 or 2 can be switched in.
Switch	V14-945101 dated 10/25/62	S9	Inverter 3 to bus 1 on-off switch	Fails closed on position		Loss of switch	With inverter 3 on ac, power cannot be removed from bus 1 by switch S9.	None	In the event of a short circuit, inverter 1 or 2 can be switched in.
				Fails closed off position		Loss of switch	No ac power available to bus 1 while inverter 3 is on.	None	Inverter 1 or 2 can be switched in.
Switch	V14-945101 dated 10/25/62	S11	Fuel cell 1 to main bus A on-off switch	Fails closed on position		Loss of switch	Fuel cell 1 dc power cannot be removed from dc main bus A by switch S11.	None	None
				Fails closed off position		Loss of switch	No dc power available to dc main bus A from fuel cell 1	None	Switch to bus B
Switch	V14-945101 dated 10/25/62	S10	Fuel cell 1 to main bus B on-off switch	Fails closed on position		Loss of switch	Fuel cell 1 dc power cannot be removed from dc main bus B by switch S10.	None	None
				Fails closed off position		Loss of switch	No dc power available to dc main bus B from fuel cell 1.	None	Switch to bus A
Switch	V14-945101 dated 10/25/62	S13	Fuel cell 2 to main bus A on-off switch	Fails closed on position		Loss of switch	Fuel cell 2 dc power cannot be removed from dc main bus A	None	None

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Switch (Cont)				Fails closed off position		Loss of switch	No dc power available to dc main bus A from fuel cell	None	Switch to bus B
Switch	V14-945101 dated 10/25/62	S12	Fuel cell 2 to main bus B on-off switch	Fails closed on position		Loss of switch	Fuel cell 2 dc power cannot be removed from dc main bus B	None	None
				Fails closed off position		Loss of switch	No dc power available to dc main bus B from fuel cell 2	None	Switch to bus A
Switch	V14-945101 dated 10/25/62	S15	Fuel cell 3 to main bus A on-off switch	Fails closed on position		Loss of switch	Fuel cell 3 dc power cannot be removed from dc main bus A.	None	None
				Fails closed off position		Loss of switch	No dc power available to dc main bus A from fuel cell 3.	None	Switch to bus B
Switch	V14-945101 dated 10/25/62	S14	Fuel cell 3 to main bus B on-off switch	Fails closed on position		Loss of switch	Fuel cell 3 dc power cannot be removed from dc main bus B.	None	None
				Fails closed off position		Loss of switch	No dc power available to dc main bus B from fuel cell 3.	None	Switch to bus A
Circuit breaker	V14-945101 dated 10/25/26	CB55	Fuel cell 1 to main bus A circuit breaker	Fails open		Loss of circuit breaker	No dc power available to main bus A from fuel cell 1.	None	Main bus B can be switched in.
				Fails closed		Loss of circuit breaker	No short circuit protection for fuel cell 1 from main bus A.	None	Main bus B can be switched in if a short occurs.

Table 15. Electrical Power System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Circuit breaker		CB56	Fuel cell 1 to main bus B circuit breaker	Fails open		Loss of circuit breaker	No dc power available to main bus B from fuel cell 1.	None	Main bus A can be switched in.
				Fails closed		Loss of circuit breaker	No short circuit protection for fuel cell 1 from main bus B	None	Main bus A can be switched in if a short occurs.
Circuit breaker	V14-945101 dated 10/25/62	CB57	Fuel cell 2 to main bus A circuit breaker	Fails open		Loss of circuit breaker	No dc power available to main bus A from fuel cell 2.	None	Main bus B can be switched in.
				Fails closed		Loss of circuit breaker	No short circuit protection for fuel cell 2	None	Main bus B can be switched in if a short occurs.

ENVIRONMENTAL CONTROL SUBSYSTEM

The major reliability engineering effort expended on the environmental control subsystem (ECS) during the last quarter consisted of (1) a prediction of ECS reliability using state-of-the-art failure rates for the various components and (2) an evaluation of subsystem modifications proposed as a means of effecting an increase in ECS reliability. Table 16 presents a summary of the predictions for the reliability of the present configuration (ECS schematic "E" change).

Table 16. Environmental Control Subsystem Reliability Requirements and Predictions

Environmental Control Subsystem Circuits	Required Reliability	Predicted Reliability	Required Crew Safety	Predicted Crew Safety
	0.997675	0.9805	0.9997	0.9996
Suit circuit	0.99949	0.9910	999931	0.99985
H_2O - glycol circuit	0.999133	0.9919	999885	0.99987
Pressure and temperature circuit	0.999812	0.99932	9999724	0.99997
Oxygen supply	0.999884	0.99957	9999816	0.99998
Water supply	0.99949	0.99834	999931	0.99993
ECS radiators*	0.99995		999998	
Cold plates*	0.99995		999999	
Waste management*	0.999965		999998	

*Apportioned values - design not finalized

The above results indicate that the greatest improvement is required in the suit and water-glycol circuits.

A redundancy study of suit compressors and glycol pumps further indicates that by adding a third fan, a compressor, and a pump (dual redundancy), or by utilizing two interchangeable cabin fans for the two-suit compressors the required reliability could be met. This is based on predicted state-of-the-art failure rate for rotating components of 40×10^{-6}.

CONFIGURATION ANALYSIS

Subsystem Description

Seven different system combinations of rotating equipment were analyzed for their probability of mission success. The abort criterion for the analysis was that an abort condition prevailed on any failure mode for which the next failure could result in catastrophic loss of the spacecraft and/or crew. The combination of rotating components and the mission success reliability equation for the rotating elements for each analyzed combination are as follows:

System I - 2 pumps, 2 compressors, 2 fans. (This is the current combination indicated on "E" revision of the ECS schematic.)

Mission Success

$$R_{ms} = \left[R^2_{1\ pump}\, R^2_{1\ comp}\right] \left[1 - (1 - R_{fan})^2\right] \left[1 - (1 - R_{2\ pump})^2\right] \left[1 - (-R_{2\ comp})^2\right]$$

System II - 2 pumps, 2 compressors, 2 fans (interchangeable with compressors)

Mission Success

$$R_{ms} = \left[R^2_{1\ pump}\right] \left[1 - (1 - R_{2\ pump})^2\right] \left\{\left[R^4_{1\ comp}\right] \left[1 - (1 - R_{2\ comp})^4\right] + \left[4\, R^3_{1\ comp}\,(1 - R_{1\ comp})\right] \left[1 - (1 - R_{2\ comp})^3\right] + \left[5\, R^2_{1\ comp}\,(1 - R_{1\ comp})^2\right] \cdot \left[1 - (1 - R_{2\ comp})^2\right]\right\}$$

System III - 2 pumps, 3 compressors, 2 fans

Mission Success

$$R_{ms} = \left[R^2_{1\ pump}\right]\left[1 - (R_{2\ pump})^2\right] \left\{\left[R^3_{1\ comp}\right]\left[1 - (1 - R_{2\ comp})^3\right] + \left[3R^2_{1\ comp}(1 - R_{1\ comp})\right]\left[1 - (1 - R_{2\ comp})^2\right]\right\} \left[1 - (1 - R_{fan})^2\right]$$

System IV - 3 pumps, 2 compressors, 2 fans

$$R_{ms} = \left[R^2_{1\ comp}\right]\left[1 - (1 - R_{2\ comp})^2\right]\left[1 - (1 - R_{3\ fan})^2\right] \left\{\left[R^3_{1\ pump}\right]\left[1 - (1 - R_{2\ pump})^3\right] + \left[3R^2_{1\ pump}(1 - R_{1\ pump})\right]\left[1 - (1 - R_{2\ pump})^2\right]\right\}$$

System V - 3 pumps, 2 compressors, 2 fans (interchangeable with compressors)

Mission Success

$$R_{ms} = \left\{\left[R^3_{1\ pump}\right]\left[1 - (1 - R_{2\ pump})^3\right] + \left[3R_{1\ pump}(1 - R_{1\ pump})\right]\left[1 - (1 - R_{2\ pump})^2\right]\right\} \left\{\left[R^4_{1\ comp}\right]\left[1 - (1 - R_{2\ comp})^4\right] + \left[4R^3_{1\ comp}(1 - R_{1\ comp})\right]\cdot\left[1 - (1 - R_{2\ comp})^3\right] + \left[5R^2_{1\ comp}(1 - R_{1\ comp})^2\right]\left[1 - (1 - R_{comp})^2\right]\right\}$$

System VI - 3 pumps, 3 compressors, 3 fans

Mission Success

$$R_{ms} = \left\{\left[R^3_{1\ pump}\right]\left[1-(1-R_{2\ pump})^3\right] + \left[3R^2_{1\ pump}(1-R_{1\ pump})\right]\left[1-(1-R_{2\ pump})^2\right]\right\}\left\{\left[R^3_{1\ comp}\right]\left[1-(1-R_{2\ comp})^3\right] + \left[3R^2_{1\ comp}(1-R_{1\ comp}\right]\left[1-(1-R_{1\ comp})^2\right]\right\}\left[1-(1-R_{3\ fan})^2\right]$$

System VII - 3 pumps, 3 compressors, 3 fans

Mission Success

$$R_{ms} = \left\{\left[R^3_{1\ pump}\right]\left[1-(1-R_{2\ pump})^3\right] + \left[3R^2_{1\ pump}(1-R_{1\ pump})\right]\left[1-(1-R_{2\ pump})^2\right]\right\}\left\{\left[R^3_{1\ comp}\right]\left[1-(1-R_{2\ comp})^3\right] + \left[3R^2_{1\ comp}(1-R_{1\ comp})\right]\left[1-(1-R_{2\ comp})^2\right]\right\}\left[1-(1-R_{3\ fan})^2\right]$$

Crew safety reliability for the foregoing seven rotating component combinations is based on evaluation of the derived probability density function shown in the Abort Criteria section.

Table 17 presents the results for the seven system variations studied. Briefly, the numerical results indicate the following:

1. The numerical mission success and crew safety probabilities for the various rotating components combinations are tabulated in columns (1) and (2).

2. By combining the results of columns (1) and (2) with the mission success and crew safety values, based on the state-of-the-art failure rates for balance of the components comprising the ECS (as presented later in this section), the over-all ECS mission success and crew safety probabilities are obtained and presented in columns (3) and (4).

Table 17. Redundant Component Trade-Off Study

Configuration				Component Reliability		ECS Reliability		S/C Reliability	
	Pumps	pressors	Fans	(1) MS	(2) CS	(3) MS	(4) CS	(5) MS	(6) CS
I	2	2	2	0.986540	0.999910	0.9805	0.99961	0.9476	0.99911
II	2	2	2*	0.993255	0.999954	0.9874	0.99965	0.9541	0.99915
III	2	3	2	0.993189	0.999954	0.9873	0.99965	0.9540	0.99915
IV	3	2	2	0.993189	0.999954	0.9873	0.99965	0.9540	0.99915
V	3	2	2*	0.999983	0.999998	0.9941	0.9997	0.9600	0.9992
VI	3	3	2	0.999883	0.999997	0.9940	0.9997	0.9600	0.9992
VII	3	3	3	0.999927	0.999997	0.9940	0.9997	0.9600	0.9992

* Redundant to compressors
ECS Environmental Control System
MS Mission success
CS Crew Safety
S/C Spacecraft

3. It may be noted that the only rotating equipment combinations that comply with the required ECS mission success and crew safety requirements are V, VI, VII.

4. The effects on spacecraft mission success and crew safety for the various rotating equipment combinations are presented in columns (5) and (6). Again, only combinations V, VI, and VII comply with the apportioned requirements.

The balance of this section is concerned with the discussion of the remainder of the ECS components and associated subsystems.

The latest ECS system schematic illustrating the "E" revision of the ECS is included as Figure 16.

The ECS is composed of the five functional subsystems included in Figure 16, and in addition, three associated subsystems are tied to and function as part of the ECS. The eight functional subsystems are described as follows:

1. Pressure suit subsystem - The purpose of the pressure suit circuit is to automatically control the flow, pressure, temperature, and composition of the pressure suit gas. The subsystem, in conjunction with the command module pressure and temperature control subsystem, will also control the environmental conditions in the cabin when any or all the crew are out of their pressure suits. The state-of-the-art failure rates for the pressure suit circuit are tabulated in Table 18. Figure 17 presents the subsystem logic block diagram.

2. Water-glycol subsystem - The water-glycol subsystem is an intermediate heat transfer loop which permits heat to be transferred from the space vehicle interior to the space radiators. This function is accomplished by pumps, heat exchangers, valves, and controls. The state-of-the-art failure rates for the water-glycol circuit are tabulated in Table 19. Figure 18 presents the subsystem logic block diagram.

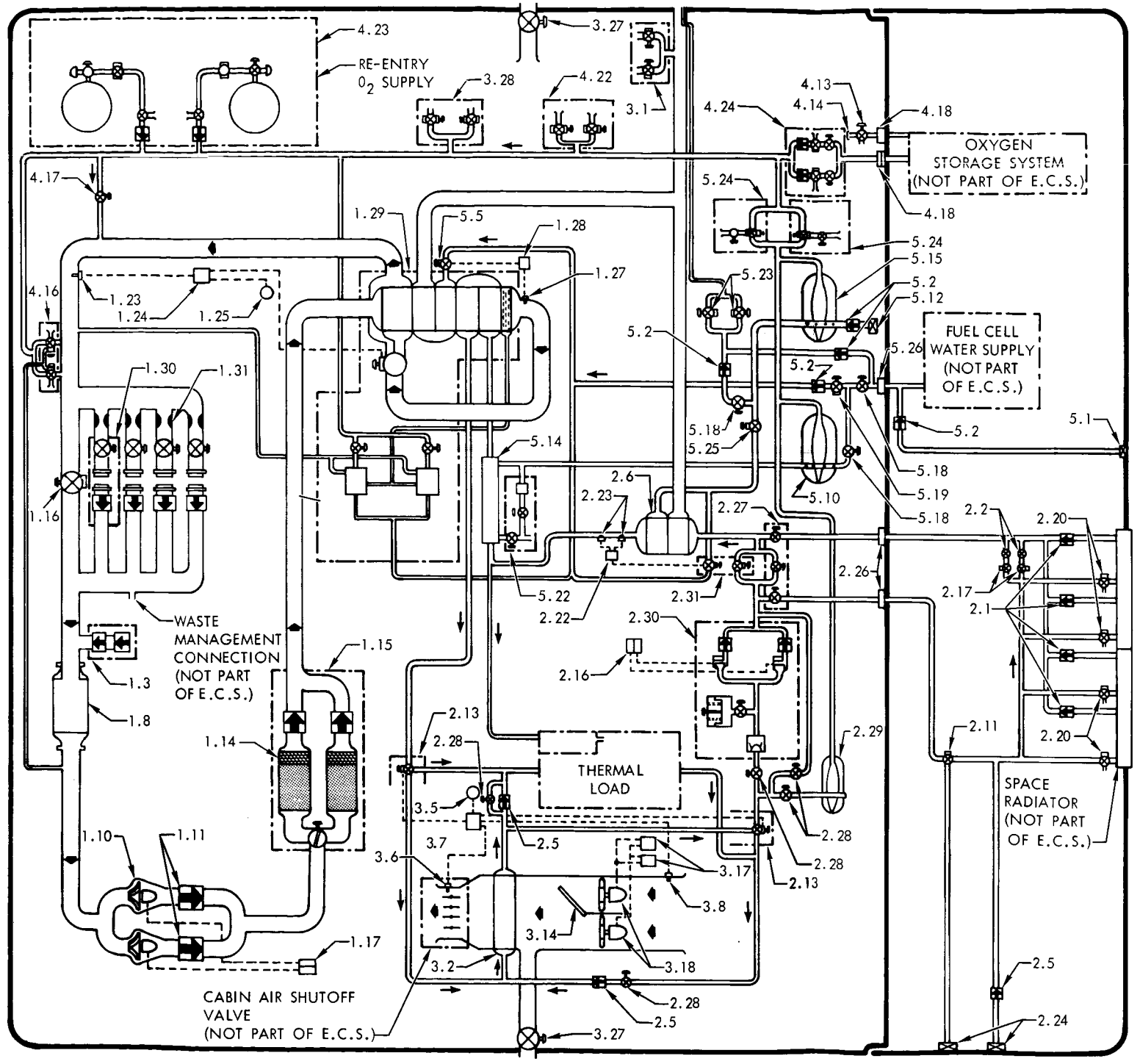
3. Command module pressure and temperature control subsystem - Although this subsystem serves several purposes, the principal function is to automatically maintain the pressure and temperature of the cabin within the prescribed limits. This function is accomplished in conjunction with the pressure suit subsystem by means of regulated oxygen inflow, recirculation blowers, a heat exchanger, a temperature control and sensors, vent valves,

Table 18. Pressure Suit Subsystem Component Reliability Values

Item	Component	Number of Components	State-of-the-Art		Operating Time (hr)
			Reliability	Failure Rate x 10^{-6}	
1.3	Suit circuit return check valve assembly	1	0.999328	2.0	336
1.8	Debris trap	1	0.999328	2.0	336
1.11	Suit compressor check valve	2	0.999328	2.0	336
1.15	CO_2 and odor absorber package				
	(a) Cannister	2	0.999664	2.0	336
	(b) Check valve	2	0.999664	1.0	
	(c) Isolation valve	1	0.999328	2.0	
1.16	Suit by-pass valve	1	0.999328	2.0	336
1.17	Unit compressor selector switch	2	0.999972	.084	
1.23	Suit air temperature sensor	1	0.999986	.042	336
1.24	Suit air temperature control	1	0.99989	.336	336
1.25	Suit air temperature selector	1	0.999978	.067	336
1.27	Suit evaporator temperature sensor	1	0.999986	.042	336
1.28	Suit evaporator temperature control	1	0.99989	.336	336
1.29	Heat Exchanger Package				
	regen. heat exchanger	1	0.999328	2.0	
	by-pass valve	1	0.999328	2.0	
	suit evaporator	1	0.999328	2.0	336
	glycol to suit heat exchanger	1	0.999328	2.0	
	water separator	1	0.999328	2.0	
	H_2O separator pump assembly	1	0.993280	4.0	
1.30	Suit hose connector package				
	supply connector	4			336
	suit return	4	0.999328	2.0	
	check valve	4			
1.31	Flow limiter and suit shut-off valve	4	0.999260	2.2	336

Table 19. Water-Glycol Subsystem Component Reliability Values

Item	Component	Number of Components	State-of-the-Art		Operating Time (hr)
			Reliability	Failure Rate x 10^{-6}	
2.1	Space radiator outlet check valve	4	0.997320	8.0	336
2.2	Glycol pressure relief valve	2	0.999328	2.0	336
2.5	Glycol check valve	3	0.999986	0.042	336
2.6	Glycol evaporator	1	0.999328	2.0	336
2.11	Glycol shutoff valve	1	0.998720	0.378	336
2.13	Cabin temperature control valve	1	0.999328	2.0	336
2.16	Glycol pump selector switch	2	0.999972	0.084	336
2.17	Glycol shutoff valve	2	0.998720	0.378	336
2.20	Space radiator isolation and vent valve	4	0.997320	8.0	336
2.22	Glycol temperature control	1	0.999890	0.336	336
2.23	Glycol temperature sensor	1	0.999986	0.042	336
2.24	Glycol fill and vent connector	2	0.999972	0.084	336
2.26	Glycol quick disconnect	2	0.999973	0.080	336
2.27	Manual shutoff valve	3	0.999328	2.0	336
2.28	Glycol shutoff valve	5	0.999323	2.0	336
2.29	Glycol reservoir	1	0.999328	2.0	336
2.30	Glycol pump assembly				
	accumulator	1	0.999328	2.0	
	isolation valve	1	0.999328	2.0	
	pump	2	0.993280	40.0	336
	check valve	2	0.999328	2.0	
	filter	1	0.9999974	0.008	
2.31	Glycol temperature control valve and diverter valve	2	0.999328	2.0	336


RE-ENTRY O_2 SUPPLY
OXYGEN STORAGE SYSTEM (NOT PART OF E.C.S.)
FUEL CELL WATER SUPPLY (NOT PART OF E.C.S.)
WASTE MANAGEMENT CONNECTION (NOT PART OF E.C.S.)
THERMAL LOAD
SPACE RADIATOR (NOT PART OF E.C.S.)
CABIN AIR SHUTOFF VALVE (NOT PART OF E.C.S.)

⊗ MANUAL SHUTOFF VALVE
⊗ ELECTRICALLY ACTUATED VALVE
⊗ ELECTRICALLY ACTUATED CONTROL VALVE WITH MANUAL OVERRIDE
⊗ PRESSURE REGULATOR
⊗ PRESSURE REGULATOR AND RELIEF VALVE
⊗ PRESSURE RELIEF VALVE
⊗ PRESSURE RELIEF VALVE WITH MANUAL OVERRIDE
⊗ ELECTRICALLY ACTUATED 3-WAY VALVE
⊗ ELECTRICALLY ACTUATED 3-WAY VALVE WITH MANUAL OVERRIDE
CHECK VALVE
⊠ SELF SEALING QUICK DISCONNECT
□ QUICK DISCONNECT

PART NO.	ITEM NO.	NO. REQUIRED	DESCRIPTION
			GROUP I - PRESSURE SUIT CIRCUIT
850022-1-1	1.3	1	SUIT CIRCUIT RETURN AIR CHECK VALVE ASSEMBLY
811100-1-1	1.8	1	DEBRIS TRAP
826000-1-1	1.10	2	SUIT COMPRESSOR
850052	1.11	2	SUIT COMPRESSOR CHECK VALVE
811450-1-1	1.14	28	CO_2 AND ODOR ABSORBER REMOVABLE CONTAINER (26 REPLACEMENTS)
811400-1-1	1.15	1	CO_2 AND ODOR ABSORBER PACKAGE LiOH CANISTER (2) CHECK VALVE (2) ISOLATION VALVE
810500-1-1	1.16	1	SUIT BYPASS VALVE
820210-1-1	1.17	2	SUIT COMPRESSOR SELECTOR SWITCH
820100-1-1	1.23	1	SUIT AIR TEMPERATURE SENSOR
820010-1-1	1.24	1	SUIT AIR TEMPERATURE CONTROL
820020-2-1	1.25	1	SUIT AIR TEMPERATURE SELECTOR
820050-1-1	1.27	1	SUIT EVAPORATOR TEMPERATURE SENSOR
820010-1-1	1.28	1	SUIT EVAPORATOR TEMPERATURE CONTROL
813000-1-1	1.29	1	HEAT EXCHANGER PACKAGE REGENERATIVE HEAT EXCHANGER REGENERATIVE HEAT EXCHANGER BYPASS VALVE SUIT EVAPORATOR GLYCOL-TO-SUIT AIR HEAT EXCHANGER WATER SEPARATOR WATER SEPARATOR PUMP ASSEMBLY
810100-1-1	1.30	4	SUIT HOSE CONNECTOR PACKAGE SUIT SHUTOFF VALVE SUIT SUPPLY CONNECTION SUIT RETURN CHECK VALVE
810600-1-1	1.31	4	SUIT FLOW LIMITER
			GROUP II - WATER-GLYCOL CIRCUIT
827030-1-1	2.1	4	SPACE RADIATOR OUTLET CHECK VALVE
827530-1-1	2.2	2	GLYCOL PRESSURE RELIEF VALVE
827030-1-1	2.5	3	GLYCOL CHECK VALVE
812000-1-1	2.6	1	GLYCOL EVAPORATOR
827570-1-1	2.11	1	GLYCOL SHUTOFF VALVE
850006-1-1	2.13	1	CABIN TEMPERATURE CONTROL VALVE
820210-1-1	2.16	2	GLYCOL PUMP SELECTOR SWITCH
827580-1-1	2.17	2	GLYCOL SHUTOFF VALVE
820220-1-1	2.20	4	SPACE RADIATOR ISOLATION AND VENT VALVE
820010-1-1	2.22	1	GLYCOL TEMPERATURE CONTROL
820080-1-1	2.23	1	GLYCOL TEMPERATURE SENSOR
827060-1-1	2.24	2	GLYCOL FILL AND VENT CONNECTION
827590-1-1	2.26	2	GLYCOL QUICK DISCONNECT
827600-1-1	2.27	1	MANUAL SHUTOFF VALVE
827050-1-1	2.28	5	GLYCOL SHUTOFF VALVE
812800-1-1	2.29	1	GLYCOL RESERVOIR
850024-1-1	2.30	1	GLYCOL PUMP ASSEMBLY ACCUMULATOR ISOLATION VALVE PUMP (2) CHECK VALVE (2) FILTER
827390-1-1	2.31	1	GLYCOL TEMPERATURE CONTROL VALVE

PART NO.	ITEM NO.	NO. REQUIRED	DESCRIPTION
			GROUP III - COMMAND MODULE PRESSURE AND TEMPERATURE CONTROL SYSTEM
810400-1-1	3.1	1	CABIN OUTFLOW, PRESSURE REGULATOR AND NEGATIVE RELIEF VALVE
812100-1-1	3.2	1	CABIN HEAT EXCHANGER
820120-1-1	3.5	1	CABIN TEMPERATURE SELECTOR
820110-1-1	3.6	1	CABIN TEMPERATURE ANTICIPATOR
820010-1-1	3.7	1	CABIN TEMPERATURE CONTROL
820100-1-1	3.8	1	CABIN TEMPERATURE SENSOR
848010-1-1	3.14	1	CABIN BLOWER DIVERTER VALVE
820210-1-1	3.17	2	CABIN BLOWER SELECTOR SWITCH
826010-1-1	3.18	2	CABIN RECIRCULATING BLOWER
850014-1-1	3.27	2	CABIN VENT VALVE
810200-1-1	3.28	1	CABIN PRESSURE REGULATOR
			GROUP IV - OXYGEN SUPPLY
827140-1-1	4.13	1	BACK PACK SUPPLY SHUTOFF AND PRESSURE RELIEF VALVE
827140-1-1	4.14	1	BACK PACK SUPPLY CAP
827300-1-1	4.16	1	DEMAND PRESSURE REGULATOR AND RELIEF VALVE
827310-1-1	4.17	1	MANUAL OXYGEN SUPPLY VALVE
827810-1-1	4.18	2	OXYGEN QUICK DISCONNECT
827550-1-1	4.22	1	EMERGENCY OXYGEN INFLOW CONTROL VALVE
827240-1-1	4.23	1	REENTRY O SUPPLY SYSTEM OXYGEN TANK (2) OXYGEN FILL PORT (2) VALVE ASSEMBLY (2) PRESSURE REGULATOR (2) RELIEF VALVE (2) CHECK VALVE (2)
827330-1-1	4.24	1	OXYGEN PRESSURE REGULATOR ASSEMBLY SHUTOFF VALVE (2) PRESSURE REGULATOR (2) RELIEF VALVE (2) CHECK VALVE (2)
			GROUP V - WATER SUPPLY SYSTEM
827520-1-1	5.1		WATER SUPPLY QUICK DISCONNECT
827360-1-1	5.2	5	WATER CHECK VALVE
827010-1-1	5.5	1	SUIT EVAPORATOR WATER INFLOW CONTROL VALVE
812300-1-1	5.10	1	POTABLE WATER TANK
827520-1-1	5.12	1	FREON QUICK DISCONNECT VALVE
812400-1-1	5.14	1	WATER CHILLER
812200-1-1	5.15	1	WASTE WATER TANK
827410-1-1	5.18	3	WATER SHUTOFF VALVE
827820-1-1	5.19	1	WATER TANK PRESSURE RELIEF VALVE
812900-1-1	5.22	1	POTABLE WATER SUPPLY ASSEMBLY SUPPLY VALVE (2) HEATER
827830-1-1	5.23	2	WATER PRESSURE RELIEF VALVE
810300-1-1	5.24	2	TANK PRESSURE CONTROL AND RELIEF VALVE
827840-1-1	5.25	1	FREON FLOW VALVE
827850-1-1	5.26	1	WATER QUICK DISCONNECT

Figure 16. Environmental Control System "E" Revision

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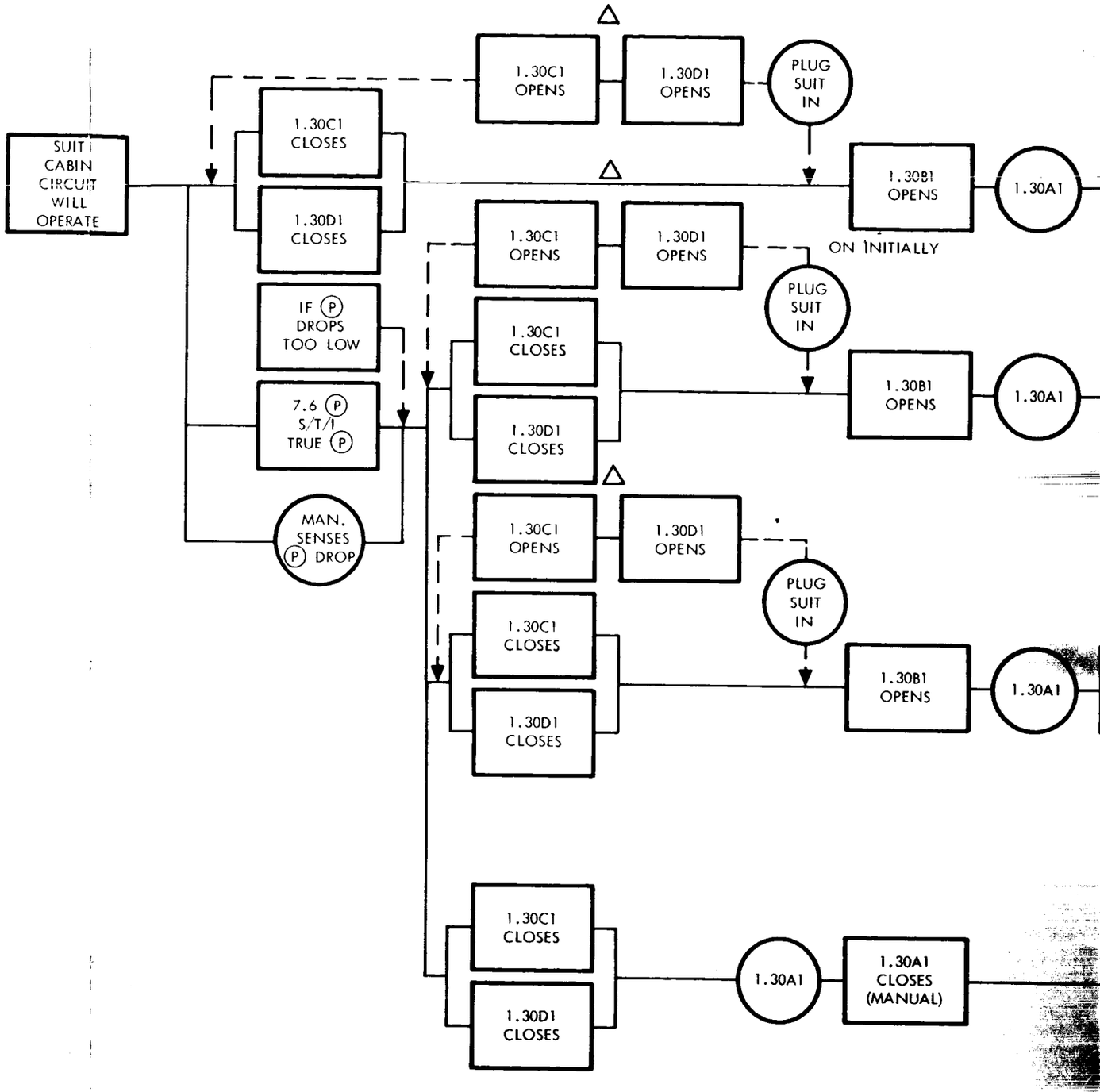
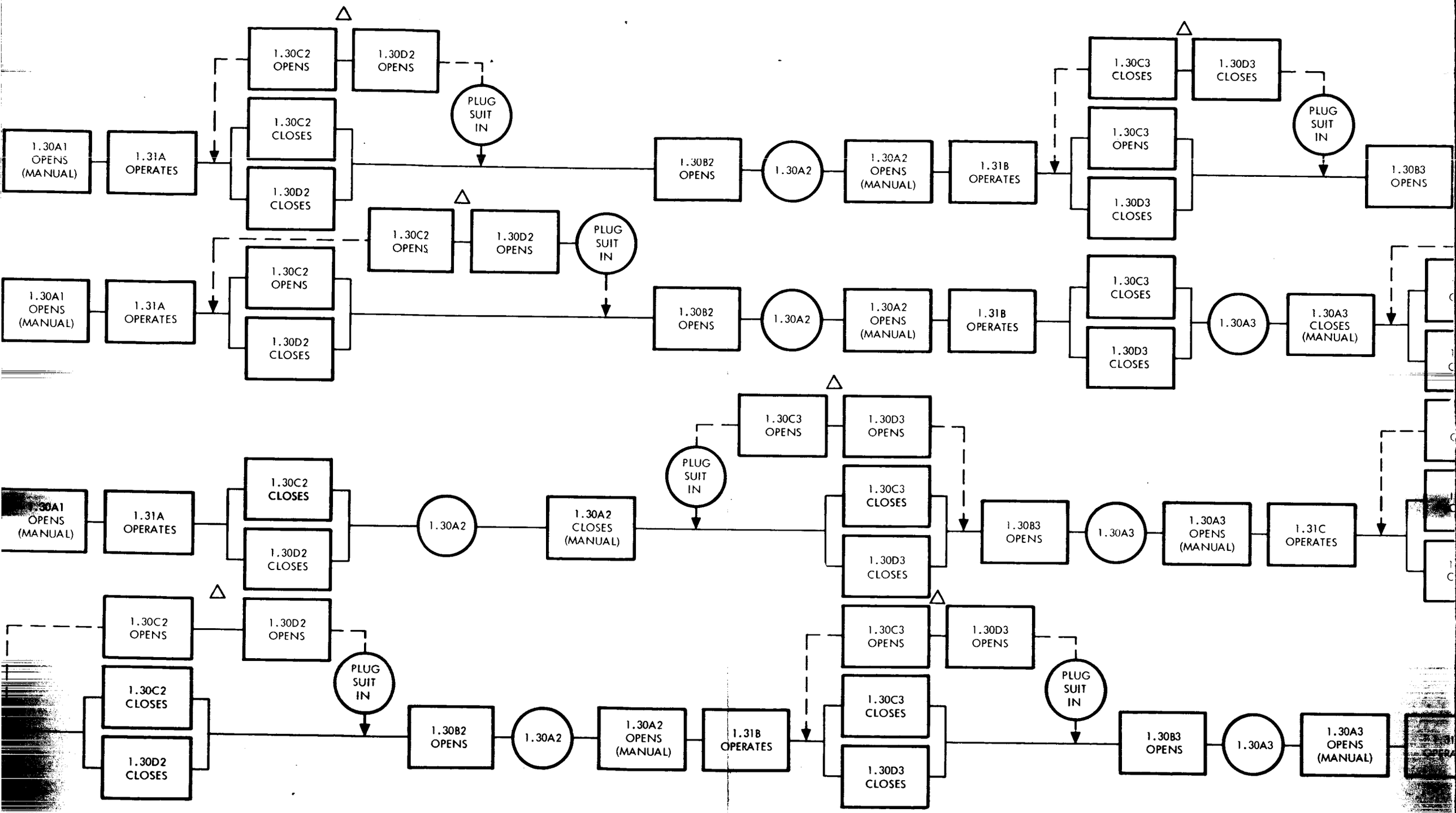
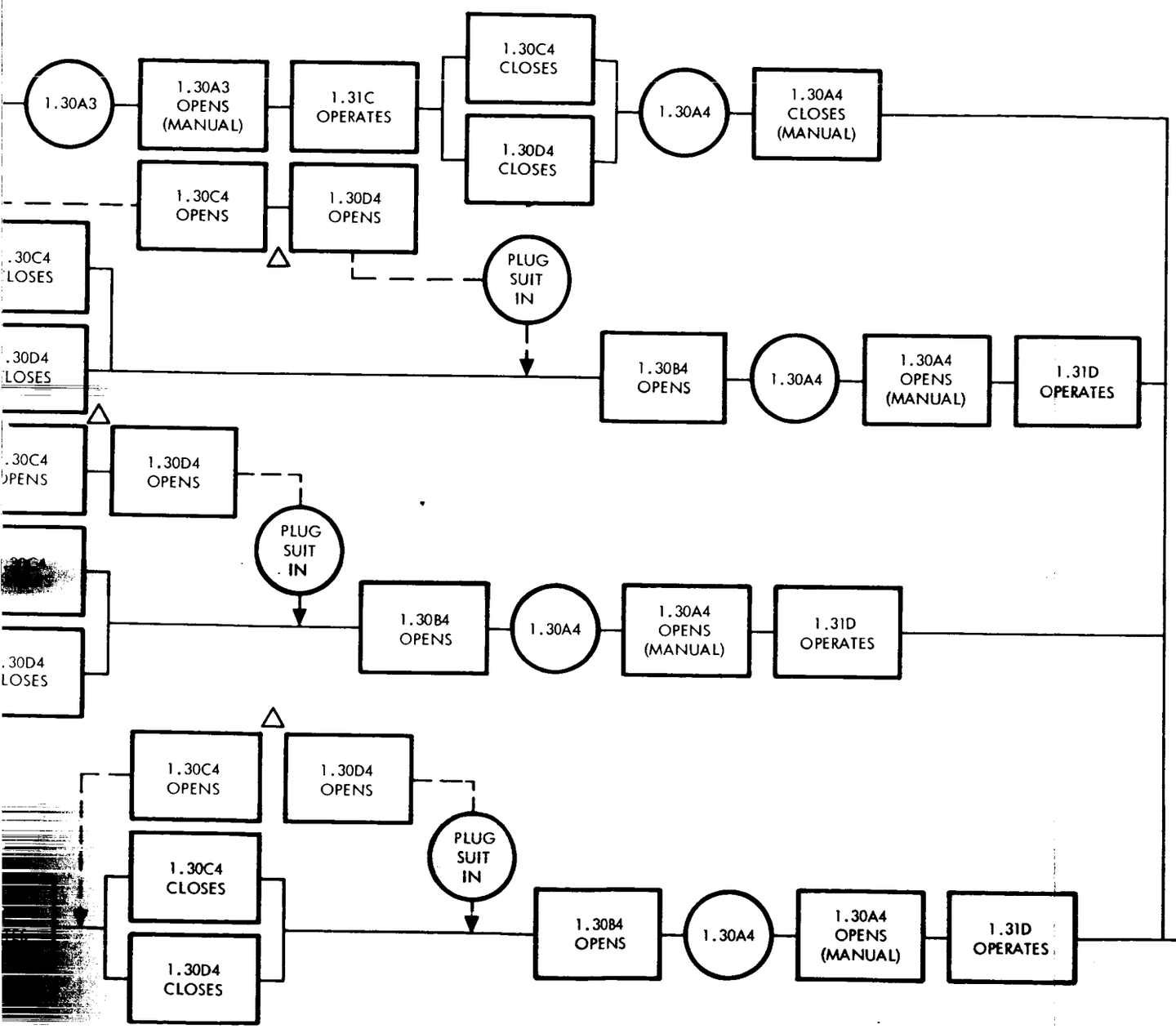
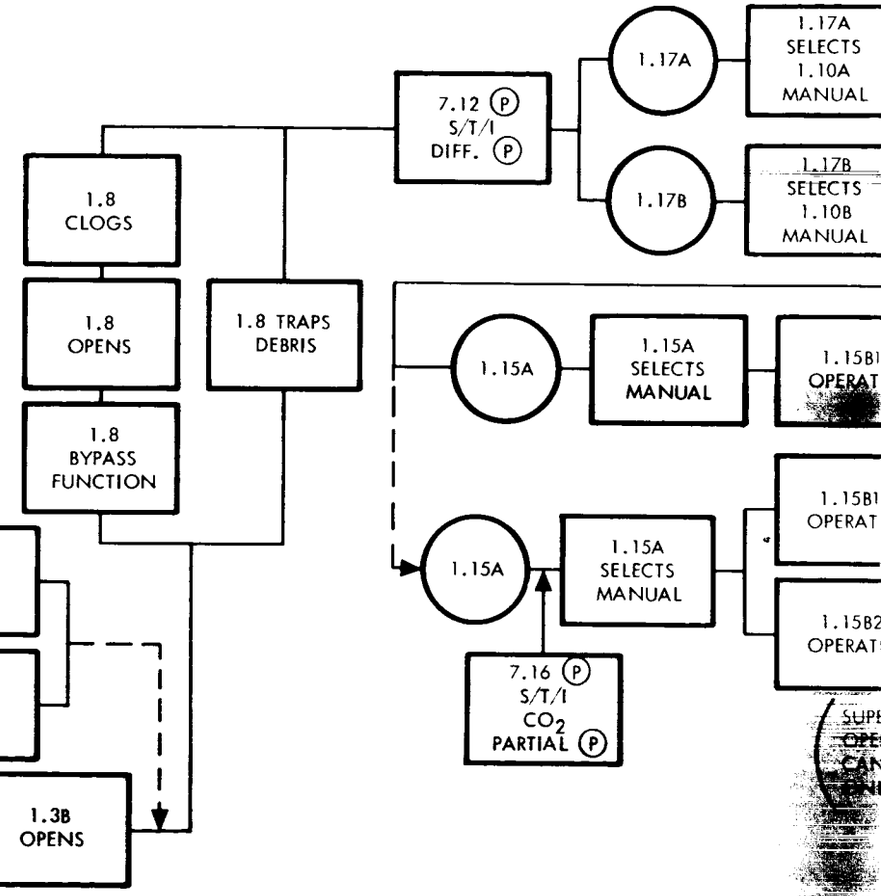


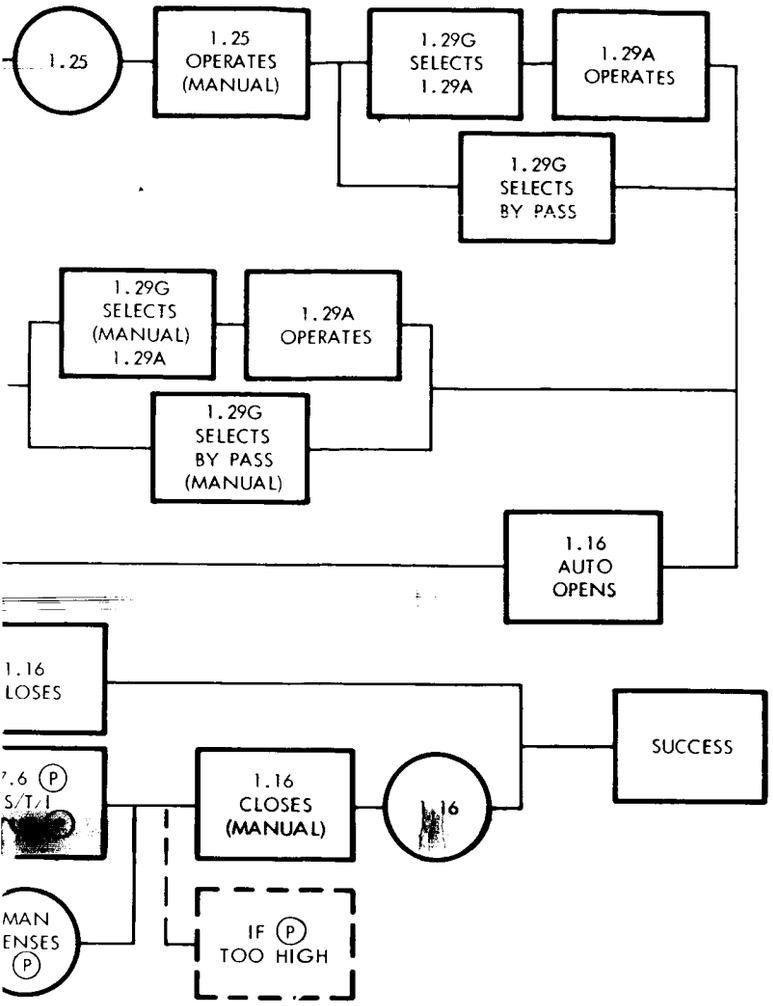

1.30A1 OPENS (MANUAL)
1.31A OPERATES
1.30C2 OPENS
1.30D2 OPENS
PLUG SUIT IN
1.30C2 CLOSES
1.30D2 CLOSES
1.30B2 OPENS
1.30A2
1.30A2 OPENS (MANUAL)
1.31B OPERATES
1.30C3 CLOSES
1.30D3 CLOSES
PLUG SUIT IN
1.30C3 OPENS
1.30D3 CLOSES
1.30B3 OPENS
1.30A1 OPENS (MANUAL)
1.31A OPERATES
1.30C2 OPENS
1.30D2 OPENS
PLUG SUIT IN
1.30C2 CLOSES
1.30D2 CLOSES
1.30B2 OPENS
1.30A2
1.30A2 OPENS (MANUAL)
1.31B OPERATES
1.30C3 CLOSES
1.30D3 CLOSES
1.30A3
1.30A3 CLOSES (MANUAL)
1.30A1 OPENS (MANUAL)
1.31A OPERATES
1.30C2 CLOSES
1.30D2 CLOSES
1.30A2
1.30A2 CLOSES (MANUAL)
PLUG SUIT IN
1.30C3 OPENS
1.30D3 OPENS
1.30C3 CLOSES
1.30D3 CLOSES
1.30B3 OPENS
1.30A3
1.30A3 OPENS (MANUAL)
1.31C OPERATES
1.30C2 OPENS
1.30D2 OPENS
PLUG SUIT IN
1.30C2 CLOSES
1.30D2 CLOSES
1.30B2 OPENS
1.30A2
1.30A2 OPENS (MANUAL)
1.31B OPERATES
1.30C3 OPENS
1.30D3 OPENS
1.30C3 CLOSES
1.30D3 CLOSES
PLUG SUIT IN
1.30B3 OPENS
1.30A3
1.30A3 OPENS (MANUAL)

2


1.30A3
1.30A3 OPENS (MANUAL)
1.31C OPERATES
1.30C4 CLOSES
1.30D4 CLOSES
1.30A4
1.30A4 CLOSES (MANUAL)
1.30C4 OPENS
1.30D4 OPENS
.30C4 LOSES
PLUG SUIT IN
.30D4 LOSES
1.30B4 OPENS
1.30A4
1.30A4 OPENS (MANUAL)
1.31D OPERATES
.30C4 PENS
1.30D4 OPENS
PLUG SUIT IN
.30D4 LOSES
1.30B4 OPENS
1.30A4
1.30A4 OPENS (MANUAL)
1.31D OPERATES
1.30C4 OPENS
1.30D4 OPENS
1.30C4 CLOSES
PLUG SUIT IN
1.30D4 CLOSES
1.30B4 OPENS
1.30A4
1.30A4 OPENS (MANUAL)
1.31D OPERATES


7.12 (P) S/T/I DIFF. (P)
1.17A
1.17A SELECTS 1.10A MANUAL
1.17B
1.17B SELECTS 1.10B MANUAL
1.8 CLOGS
1.8 OPENS
1.8 TRAPS DEBRIS
1.8 BYPASS FUNCTION
1.15A
1.15A SELECTS MANUAL
1.15A
1.15A SELECTS MANUAL
7.16 (P) S/T/I CO_2 PARTIAL (P)
1.3B CLOSES
1.3A CLOSES
1.3A OPENS
1.3B OPENS

1.11B CLOSES
1.11A OPENS
1.10A OPERATES
1.11A CLOSES
1.11B OPENS
1.10B OPERATES
1.15C1 OPENS
1.15B2 OPERATES
1.15C2 OPENS
1.15C1 OPENS
RE-PLACEM'T CANNIS-TER
1.14A REPLACEMENT CANNISTER AVAILABLE
1.15C2 CLOSES
1.15C2 OPENS
RE-PLACEM'T CANNIS-TER
1.14B REPLACEMENT CANNISTER AVAILABLE
1.15C1 CLOSES
NORMAL
DURING
CHANGE
1.29A AIR COOLS AIR
1.29B GLYCOL COOLS AIR
COOL GLYCOL AVAILABLE
COOL AIR AVAILABLE
1.27 (T) SENSES TRUE TEMP
1.28 CONTROLS 5.5 VALVE
5.5 O/C (AUTO)
1.29C OPERATES
WATER AVAILABLE
SV
OPENS (MANUAL)
MAN SENSES (T)
5.5
5.5 O/C (MANUAL)
7.3 (T) S/T/I TRUE (T)
IF (T) TOO HIGH
1.23 (T) SENSES TRUE (T)
1.24 OPERATES
MAN SENSES (T)
1.25 (D) S/T/I TRUE (D)
1.29G
1.29D SEPARATES H_2O
MAN SENSES HUM
1.29E1
1.29E2
1.29E1 O/C MANUAL
1.29E2 O/C MANUAL
1.29F1 OPERATES
1.29F2 OPERATES
OXYGEN PRESSURE AVAILABLE
OXYGEN PRESSURE AVAILABLE
5.2C O/C
5.23A RELIEVES
5.23B RELIEVES



COMPONENT NUMBER	ITEM DESCRIPTION
1.1	VALVE, CHECK, RASPBERRY, NORMALLY CLOSED
1.2	HEAT EXCHANGER, AIR-AIR REGENERATION CROSS-FLOW
1.3	VALVE, CHECK, DUAL, BUTTERFLY, NORMALLY CLOSED
1.4	VALVE, LIMITING AND MANUAL SHUTOFF, NORMALLY OPEN
1.5	CONNECTOR, NORMALLY OPEN
1.6	VALVE, CHECK, FLAPPER, NORMALLY CLOSED
1.7	CONNECTOR, ORIFICE, NORMALLY CLOSED
1.8	DEBRIS TRAP, SCREEN FILTER
1.9	CATALYTIC FILTER
1.10	COMPRESSOR, CENTRIFUGAL, 10,000 RPM
1.11	VALVE, CHECK, RASPBERRY, NORMALLY CLOSED
1.12	VALVE, MANUAL SHUTOFF, NORMALLY OPEN
1.13	VALVE, CHECK RASPBERRY, NORMALLY CLOSED
1.14	CO_2 AND ODOR ABSORBER REMOVABLE CONTAINER (26 REPLACEMENTS)
1.15	CO_2 AND ODOR ABSORBER
1.16	VALVE, PRESSURE RELIEF WITH MANUAL OVERRIDE, NORMALLY CLOSED
1.17	SWITCH, ROTARY, FOUR-POSITION ASSEMBLY
1.18	VALVE, ELECTRICALLY ACTUATED CONTROL WITH MANUAL OVERRIDE
1.19	HEAT EXCHANGER, GLYCOL-AIR CROSS-FLOW
1.20	HEAT EXCHANGER, WATER-AIR CROSS-FLOW
1.21	VALVE, DIVERTER, MANUAL
1.22	WATER SEPARATOR, WITH SHUTOFF VALVE AND ACTUATOR
1.23	SUIT AIR TEMPERATURE SENSOR
1.24	TEMPERATURE CONTROL, MAGNETIC AMPLIFIER
1.25	TEMPERATURE SELECTOR, RHEOSTAT
1.27	SUIT EVAPORATOR TEMPERATURE CONTROL
1.28	TEMPERATURE CONTROL, MAGNETIC AMPLIFIER
1.29	HEAT EXCHANGER PACKAGE REGENERATIVE HEAT EXCHANGER REGENERATIVE HEAT EXCHANGER BYPASS VALVE SUIT EVAPORATOR GLYCOL-TO-SUIT AIR HEAT EXCHANGER WATER SEPARATOR WATER SEPARATOR PUMP ASSEMBLY
1.30	SUIT HOSE CONNECTOR PACKAGE SUIT SHUTOFF VALVE SUIT SUPPLY CONNECTION CHECK VALVE
1.31	SUIT FLOW LIMITER
5.2	WATER CHECK VALVE
5.23	WATER PRESSURE RELIEF VALVE
5.5	VALVE, ELECTRICALLY ACTUATED CONTROL WITH MANUAL OVERRIDE
5.13	VALVE, CHECK, QUADRUPLE ASSEMBLY, NORMALLY OPEN
7.1	SENSOR, AIR PRESSURE
7.3	SENSOR, AIR TEMPERATURE
7.6	SENSOR, PRESSURE DIFFERENTIAL
7.7	SENSOR, PRESSURE DIFFERENTIAL
7.9	SENSOR, AIR TEMPERATURE
7.10	SENSOR, VOLTAGE INDICATION
7.11	SENSOR, AIR TEMPERATURE
7.12	TRANSDUCER, PRESSURE, INLET SUIT SUPPLY MANIFOLD
7.16	TRANSDUCER, PARTIAL PRESSURE, CO_2

LEGEND

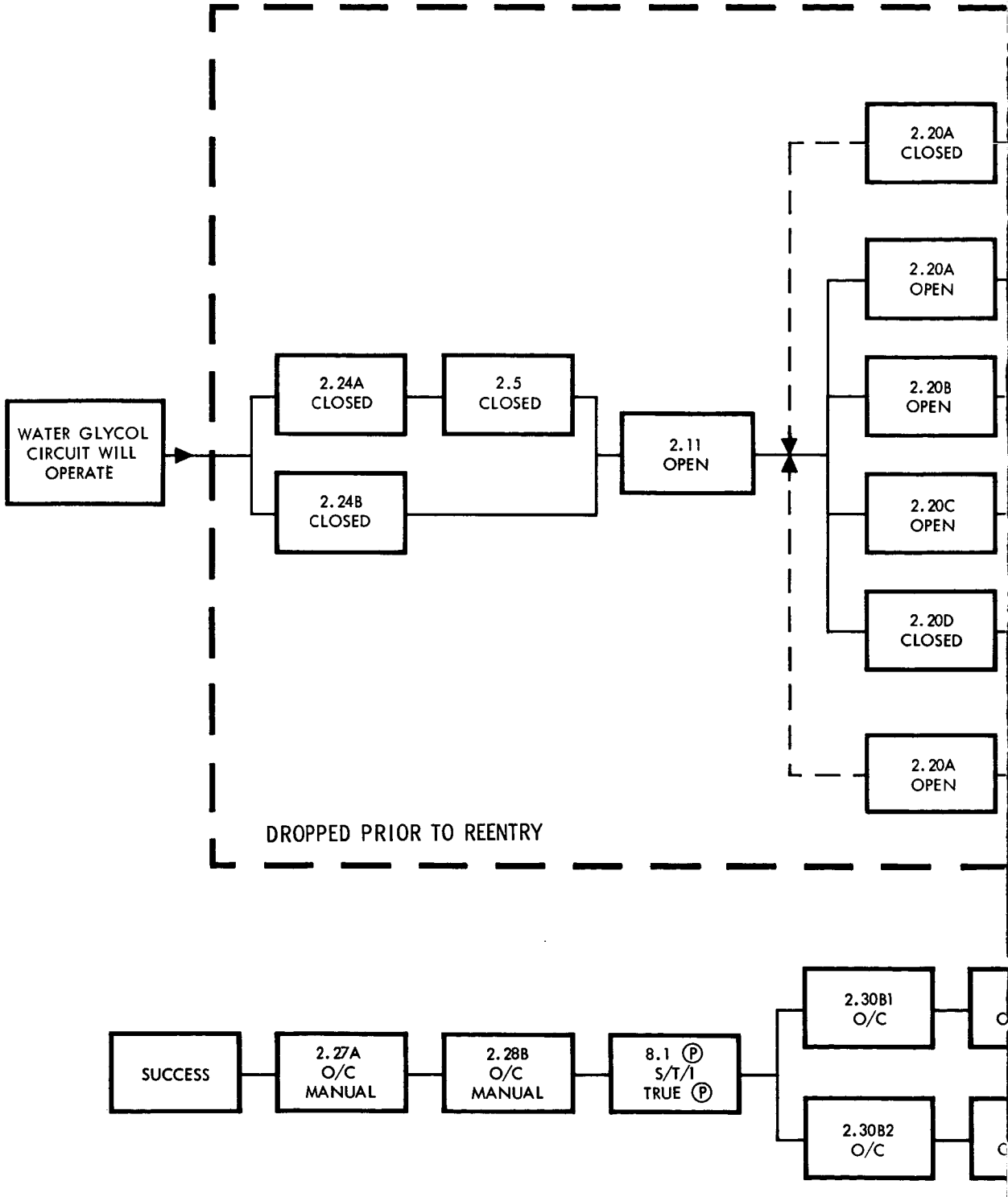
△ = CABIN EMERGENCY SUIT MODE
○ = HUMAN OPERATION
□ = MECHANICAL OPERATION
S/T/I = SENSES, TRANSMITS, AND INDICATES
S/T/W = SENSES, TRANSMITS, AND WARNS
(D) = POSITION
(P) = PRESSURE
(T) = TEMPERATURE
(V) = VOLTAGE
S/V = SERVO VALVE
O/C = OPEN OR CLOSED

NOTE:

A, B, C, ETC, AFTER COMPONENT NUMBER INDICATES POSITION IN CIRCUIT OF SUCCESSIVE IDENTICAL COMPONENT

NOTE: DOES NOT INCLUDE GROUND CHECK-OUT PROVISIONS
UNLESS OTHERWISE INDICATED PARALLEL REDUNDANCY IS ASSUMED RATHER THAN SEQUENTIAL REDUNDANCY
NORMAL OPERATION AND NORMAL CONDITIONS ASSUMED UNLESS OTHERWISE INDICATED
LOGIC BASED UPON REQUIREMENTS FOR SUCCESS

Figure 17. Pressure Suit Subsystem Logic Diagram


WATER GLYCOL CIRCUIT WILL OPERATE
2.24A CLOSED
2.24B CLOSED
2.5 CLOSED
2.11 OPEN
2.20A CLOSED
2.20A OPEN
2.20B OPEN
2.20C OPEN
2.20D CLOSED
2.20A OPEN
DROPPED PRIOR TO REENTRY

SUCCESS
2.27A O/C MANUAL
2.28B O/C MANUAL
8.1 (P) S/T/I TRUE (P)
2.30B1 O/C
2.30B2 O/C

EARTH CHECKOUT AND ASCENT

2.1A CLOSED
2.20B CLOSED
2.1B CLOSED
2.20C CLOSED
2.1C CLOSED
2.20D OPEN
2.1D CLOSED
2.17A OPEN
2.2A OPERATES
2.17B OPEN
2.2B OPERATES

SPACE RADIATOR A
2.1A OPEN
2.1B CLOSED
2.1C CLOSED
2.1D CLOSED

SPACE RADIATOR B
2.1B OPEN
2.1A CLOSED
2.1C CLOSED
2.1D CLOSED

SPACE RADIATOR C
2.1C OPEN
2.1A CLOSED
2.1B CLOSED
2.1D CLOSED

SPACE RADIATOR D
2.1D OPEN
2.1A CLOSED
2.1B CLOSED
2.1C CLOSED

2.17A O/C
2.2A OPERATES
2.17B O/C
2.2B OPERATES

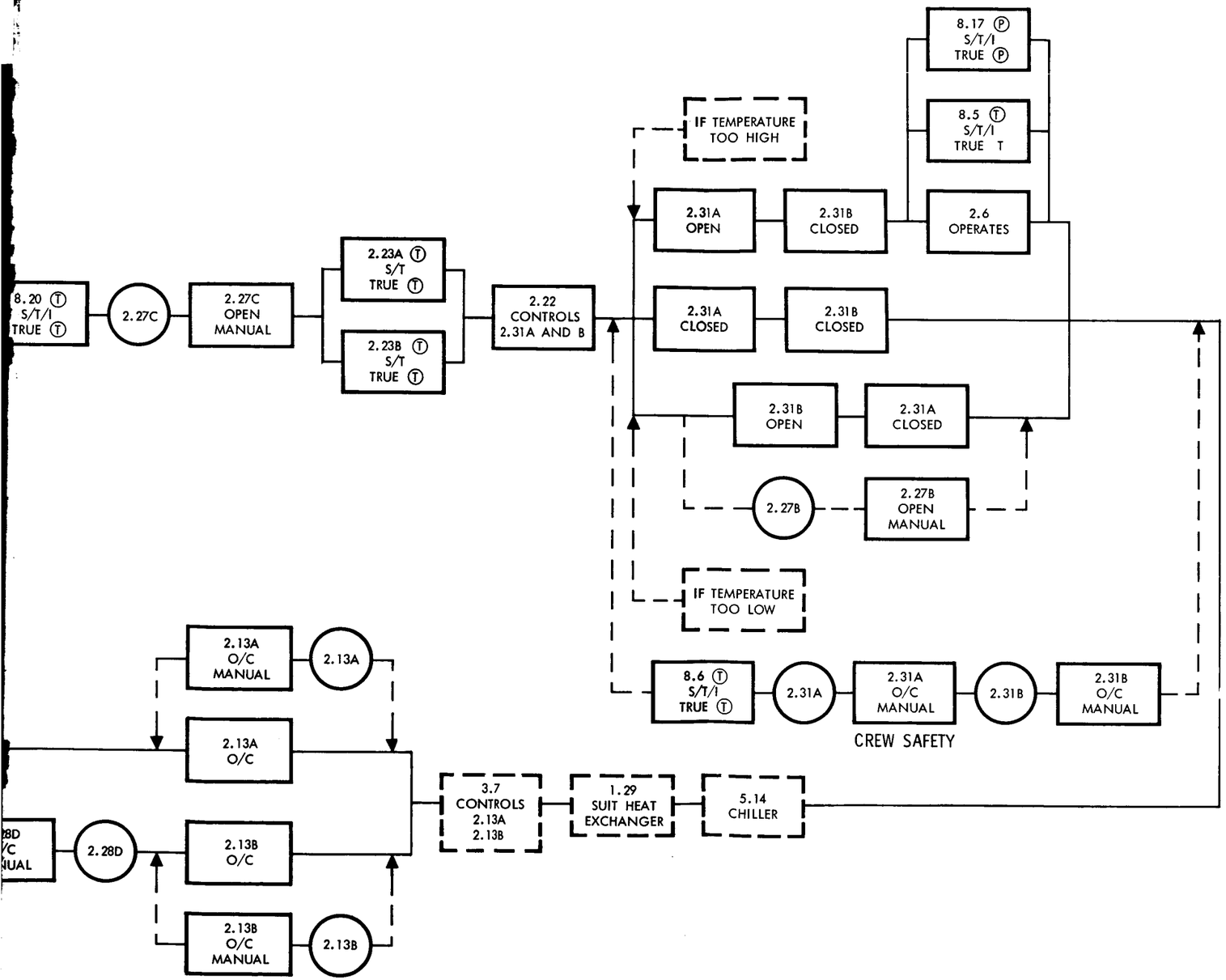
2.26A OPERATES
2.26B OPERATES

SPACE MODE

SPACE RADIATOR A
2.1A OPEN
2.20B OPEN
SPACE RADIATOR B
2.1B OPEN
2.20C OPEN
SPACE RADIATOR C
2.1C OPEN
2.20D OPEN
SPACE RADIATOR D
2.1D OPEN
2.17A CLOSED
2.17B CLOSED

MOON-SUBSOLAR MODE

2.30A1 PERATES
2.16A SELECTS 2.30A1
2.16A
2.30A2 PERATES
2.16B SELECTS 2.30A2
2.16B
8.16 S/T/I
2.30F OPERATES
2.30E O/C MANUAL
2.30D OPERATES
2.28C OPEN MANUAL
2.28C
2.28A OPEN MANUAL
2.28A
2.29 OPERATES
OXYGEN PRESSURE AVAILABLE
2.28E O/C MANUAL
2.28E
2.5B O/C
3.2 CABIN AIR
2.5A O/C



COMPONENT NUMBER	ITEM DESCRIPTION
1.29	HEAT EXCHANGER PACKAGE
	REGENERATIVE HEAT EXCHANGER
	REGENERATIVE HEAT EXCHANGER BYPASS VALVE
	SUIT EVAPORATOR
	GLYCOL-TO-AIR HEAT EXCHANGER
	WATER SEPARATOR
	WATER SEPARATOR PUMP ASSEMBLY
2.1	SPACE RADIATOR OUTLET CHECK VALVE
2.2	GLYCOL PRESSURE RELIEF VALVE
2.5	GLYCOL CHECK VALVE
2.6	GLYCOL EVAPORATOR
2.11	GLYCOL SHUTOFF VALVE
2.13	CABIN TEMPERATURE CONTROL VALVE
2.16	GLYCOL PUMP SELECTOR SWITCH
2.17	GLYCOL SHUTOFF VALVE
2.20	SPACE RADIATOR ISOLATION AND VENT VALVE
2.22	GLYCOL TEMPERATURE CONTROL
2.23	GLYCOL TEMPERATURE SENSOR
2.24	GLYCOL FILL AND VENT CONNECTION
2.26	GLYCOL QUICK DISCONNECT
2.27	MANUAL SHUTOFF VALVE
2.28	GLYCOL SHUTOFF VALVE
2.29	GLYCOL RESERVOIR
2.30	GLYCOL PUMP ASSEMBLY
	ACCUMULATOR
	ISOLATION VALVE
	PUMP (2)
	CHECK VALVE (2)
	FILTER
2.31	GLYCOL TEMPERATURE CONTROL VALVE
3.7	CABIN TEMPERATURE CONTROL
5.14	WATER CHILLER
8.1	TRANSDUCER, PRESSURE, OUTLET, GLYCOL PUMP
8.5	TRANSDUCER, TEMPERATURE, GASEOUS, OUTLET, GLYCOL EVAP. SENSOR
8.6	TRANSDUCER, TEMPERATURE, LIQUID, OUTLET, GLYCOL EVAP. SENSOR
8.16	TRANSDUCER, PRESSURE, QUANTITY MEASUREMENT, GLYCOL ACCOM.
8.17	TRANSDUCER, PRESSURE, GASEOUS, OUTLET, GLYCOL EVAPORATOR
8.20	TRANSDUCER, TEMPERATURE, OUTLET, SPACE RADIATOR SENSOR

LEGEND

- ○ = HUMAN OPERATION
- □ = MECHANICAL OPERATION
- O/C = OPEN OR CLOSED
- (P) = PRESSURE
- (Q) = QUANTITY
- S/T/I = SENSES, TRANSMITS, AND INDICATES
- (T) = TEMPERATURE

NOTE:
A, B, C, ETC., AFTER COMPONENT NUMBER INDICATES POSITION IN CIRCUIT OF SUCCESSIVE INDENTICAL COMPONENT

Figure 18. Water-Glycol Circuit Logic Diagram

and other valves and controls as required. State-of-the-art failure rates for the command module pressure and temperature control subsystem are tabulated in Table 20. Figure 19 presents the subsystem logic block diagram.

4. Oxygen supply subsystem - The oxygen supply subsystem is subdivided further into three subsystems: the normal oxygen supply system, the reentry oxygen supply system, and the back pack oxygen supply system. The normal oxygen supply system is capable of controlling the inflow of oxygen for the entire Apollo mission up to the reentry phase. This function is accomplished by suitable pressure regulators, demand regulators, and shutoff and check valves. The reentry oxygen supply subsystem supplies the oxygen required for mission completion after separation of the service module from the command module. This function is accomplished by oxygen storage tanks, pressure regulators, and valves. The back pack oxygen supply system supplies the oxygen required for changing the back packs within the command module from the main oxygen tanks. This function is accomplished by a shut-off and pressure relief valve. State-of-the-art failure rates for the oxygen supply subsystem are tabulated in Table 21. Figure 20 presents the subsystem logic block diagram, and Figure 21 presents the back pack supply.

5. Water supply subsystem - The water supply subsystem collects, stores, and provides water for supplemental and emergency heat transfer operation and collects and stores potable water for consumption by the crew members. The above functions are accomplished, utilizing water tanks, pressure controls, and valves. This subsystem also contains several valves associated with other subsystems. State-of-the-art failure rates for the water supply subsystem are tabulated in Table 22. Figure 22 presents the subsystem logic block diagram. As an aid to performing an ECS system reliability analysis, logic diagrams, Figures 17 through 22, were prepared for each ECS circuit. In addition, a failure mode and effects analysis for each ECS component is herein included. From these logic diagrams and effects analyses, the predicted state-of-the-art failure rates, and a 336-hour mission time, the predicted mission success and crew safety numerics presented were computed.

6. ECS radiators - The ECS radiators, although not considered part of the ECS, are connected to the water-glycol subsystem. The radiators provide the means of transferring to space the heat absorbed by the water-glycol. Two radiators are provided, which

Table 20. Pressure and Temperature Control Component Reliability Values

Item	Component	Number of Compo-nents	Reliability	Failure Rate x 10^{-6}	Operating Time (hr)
3.1	Cabine outflow pressure regulator and negative relief valve	1	0.996640	20.0	336
3.5	Cabin temperature selector	1	0.999979	0.067	336
3.2	Cabin heat exchanger	1	0.999528	3.0	336
3.6	Cabin temperature anticipator	1	0.999986	0.042	336
3.7	Cabin temperature control	1	0.999890	0.336	336
3.8	Cabin temperature sensor	1	0.999986	0.042	336
3.14	Cabin blower diverter valve	1	0.999328	2.0	336
3.17	Cabin blower selector switch	2	0.999972	0.084	336
3.18	Cabin recirculating blower	2	0.993280	40.0	336
3.27	Cabin vent valve	2	0.999328	2.0	336
3.28	Cabin pressure regulator	1	0.996640	20.0	336

Table 21. Oxygen Regulator Subsystem and Back Pack Component Reliability Values

Item	Component	Number of Compo-nents	State-of-the-Art		Operating Time (hr)
			Reliability	Failure Rate x 10^{-6}	
4.13	Back pack supply shutoff and pressure relief valve	1	0.998600	4.0	336
4.14	Back pack supply cap	1	0.9999974	0.008	336
4.16	Demand pressure regulator and relief valve	1	0.996640	20.0	336
4.17	Manual Oxygen supply valve	1	0.999328	2.0	336
4.18	Oxygen quick disconnect	2	0.999328	2.0	336
4.22	Emergency Oxygen inflow control valve	1	0.996640	20.0	336
4.23	Reentry Oxygen supply				
	Oxygen tank	2	0.9999943	0.017	
	Oxygen fill port	2	0.999972	0.084	
	valve assembly	2	0.999972	0.084	336
	pressure regulator	2	0.999930	0.210	
	relief valve	2	0.999972	0.084	
	check valve	2	0.999986	0.042	
4.24	Oxygen pressure regulator assembly				
	shutoff valve	2	0.999986	0.042	
	pressure regulator	2	0.999944	0.168	336
	relief valve	2	0.999972	0.084	
	check valve	2	0.9999916	0.025	

Table 22. Water Regulation Subsystem Component Reliability Values

Item	Component	Number of Compo-nents	State-of-the-Art		Operating Time (hr)
			Reliability	Failure Rate x 10^{-6}	
5.1	Water supply quick disconnect	1	0.999328	2.0	336
5.2	Water check valve	5	0.999328	2.0	336
5.5	Suit evaporator H_2O inflow control valve	1	0.999328	2.0	336
5.10	Portable water tank	1	0.999328	2.0	336
5.12	Freon disconnect valve	1	0.999328	2.0	336
5.14	Water chiller	1	0.999328	2.0	336
5.15	Waste water tank	1	0.999328	2.0	336
5.18	Water shut-off valve	3	0.999986	0.042	336
5.19	Water tank pressure relief valve	1	0.999972	0.084	336
5.22	Portable water supply assembly				
	Supply valve	2	0.998600	4.0	336
	Heater	1	0.999328	2.0	336
5.23	Water pressure relief valve	2	0.999972	0.084	336
5.24	Tank pressure control and relief valve	2	0.9999983	0.005	336
5.25	Freon flow valve	1	0.999328	2.0	336
5.26	Water quick disconnect	1	0.999328	2.0	336

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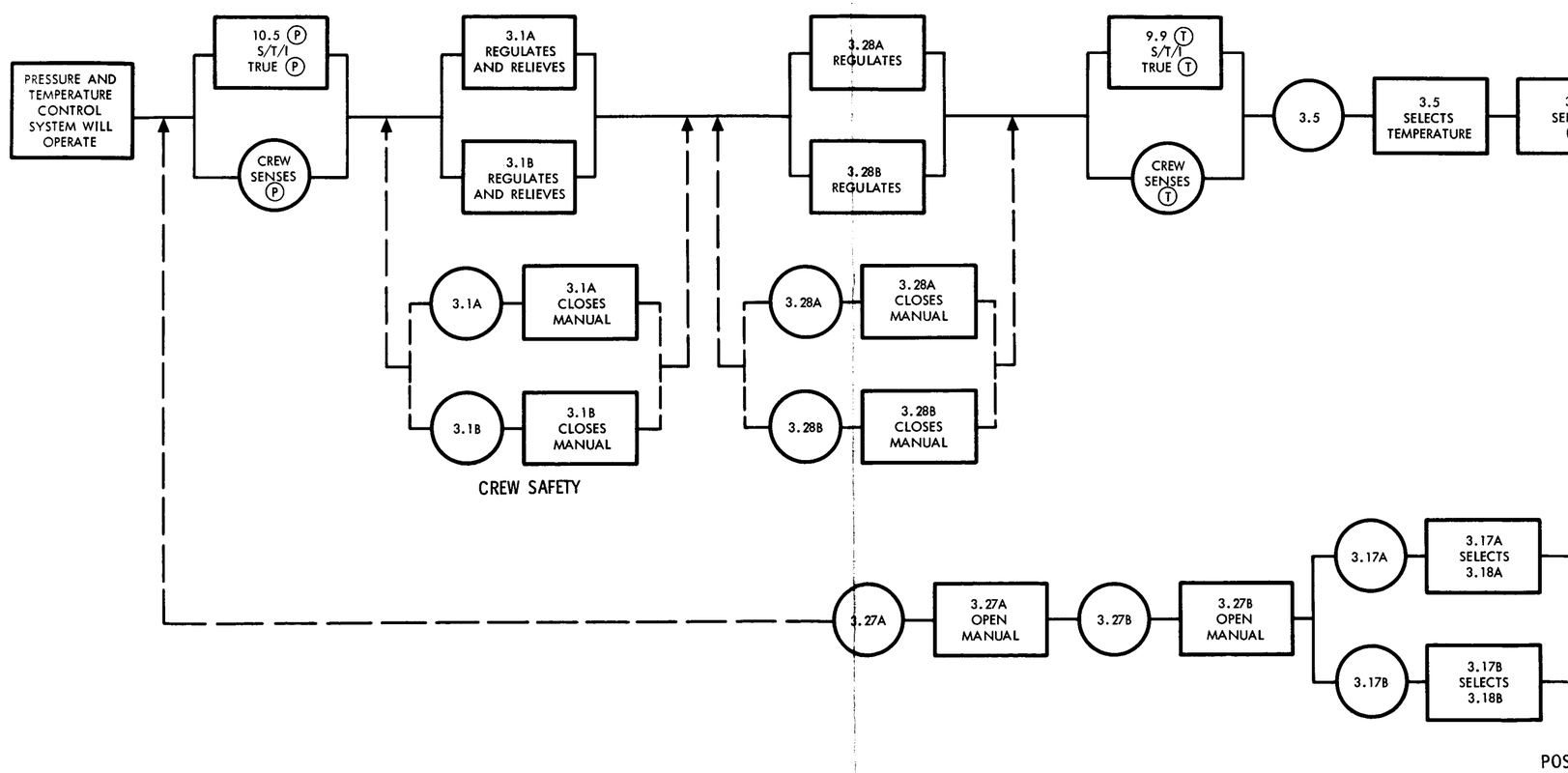
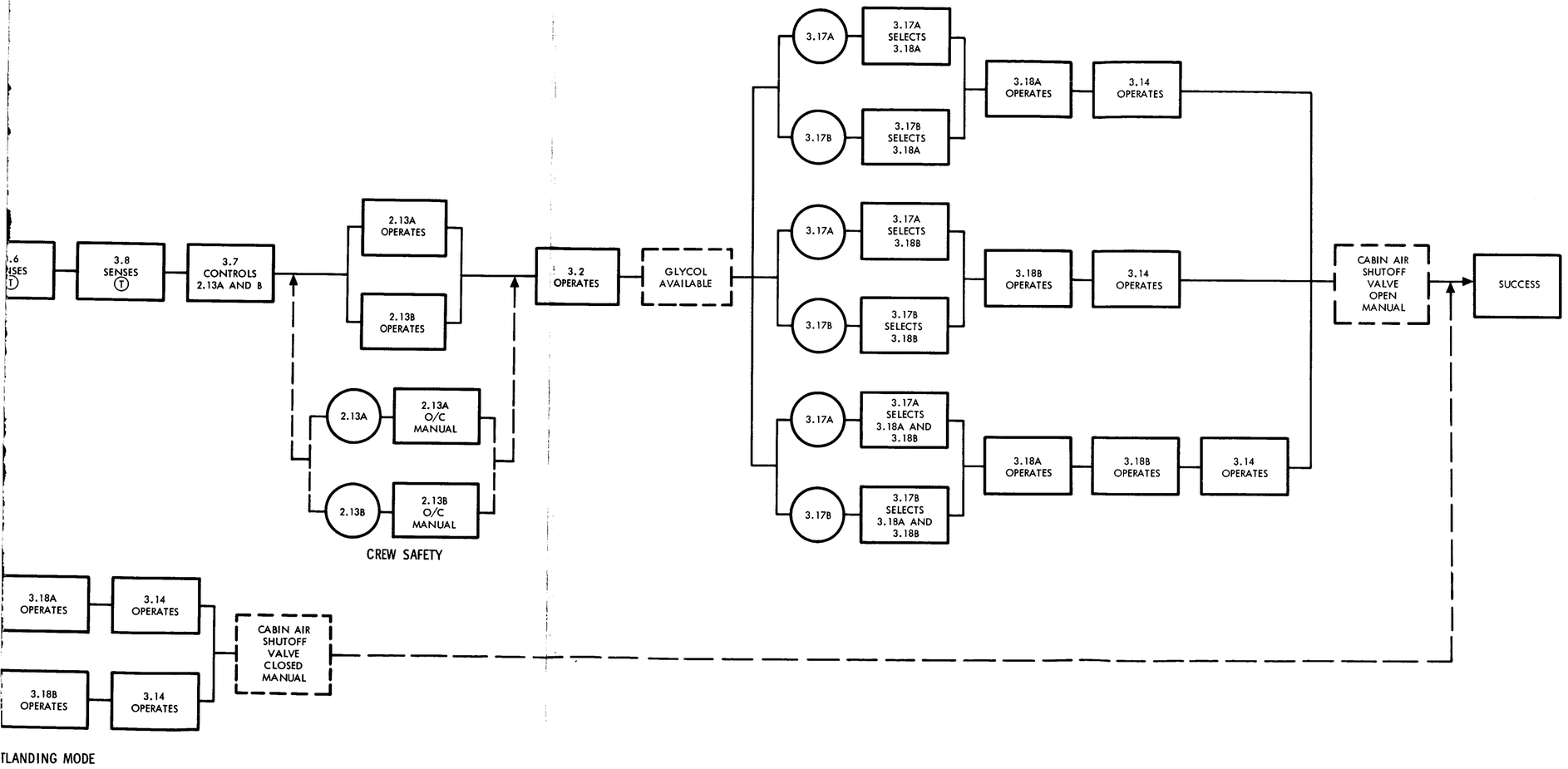




COMPONENT NUMBER	ITEM DESCRIPTION
2.13	CABIN TEMPERATURE CONTROL VALVE
3.1	CABIN OUTFLOW, PRESSURE REGULATOR AND NEGATIVE RELIEF VALVE
3.2	CABIN HEAT EXCHANGER
3.5	CABIN TEMPERATURE SELECTOR
3.6	CABIN TEMPERATURE ANTICIPATOR
3.7	CABIN TEMPERATURE CONTROL
3.8	CABIN TEMPERATURE SENSOR
3.14	CABIN BLOWER DIVERTER VALVE
3.17	CABIN BLOWER SELECTOR SWITCH
3.18	CABIN RECIRCULATING BLOWER
3.27	CABIN VENT VALVE
3.28	CABIN PRESSURE REGULATOR
9.9	TRANSDUCER, TEMPERATURE, COMMAND MODULE, AIR, CABIN SENSOR
10.5	SENSOR, COMMAND MODULE TOTAL PRESSURE

LEGEND

○ = HUMAN OPERATION

▭ = MECHANICAL OPERATION

O/C = OPEN OR CLOSED

Ⓟ = PRESSURE

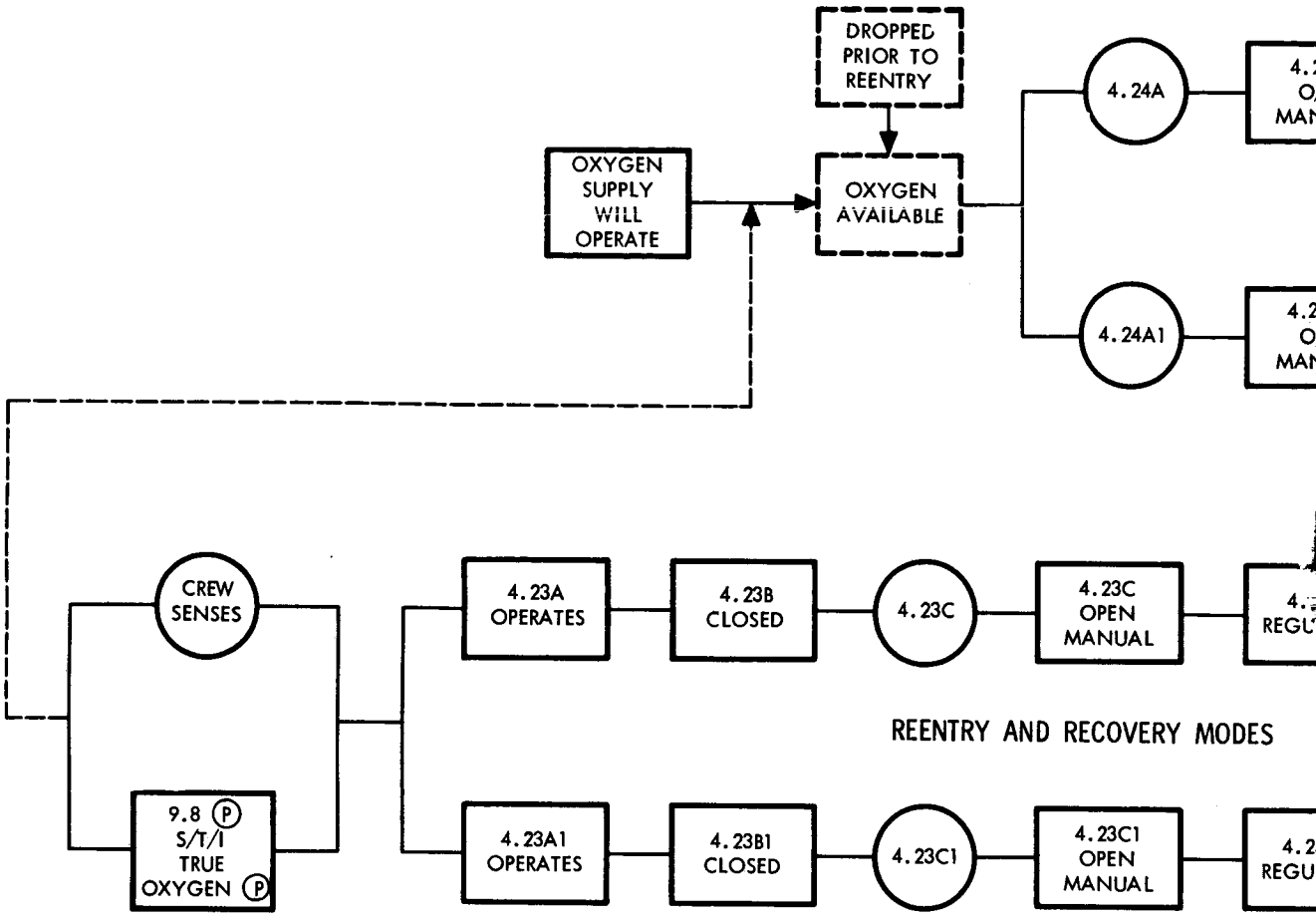
S/T/I = SENSES, TRANSMITS, AND INDICATES

Ⓣ = TEMPERATURE

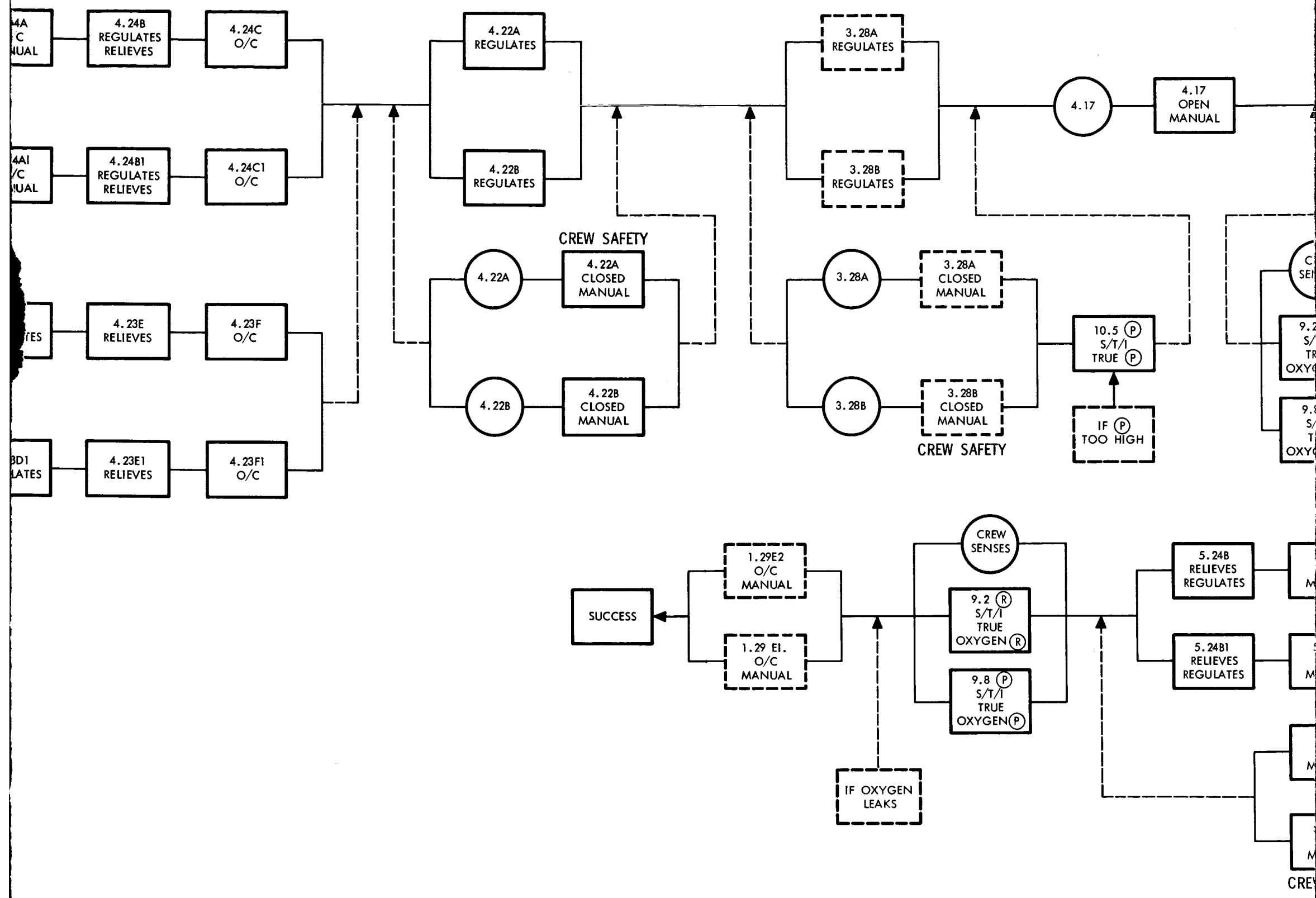
NOTE:
A, B, C, ETC., AFTER COMPONENT NUMBER INDICATES POSITION IN CIRCUIT OF SUCCESSIVE IDENTICAL COMPONENT

I. D. E 4
SID 62-557-4

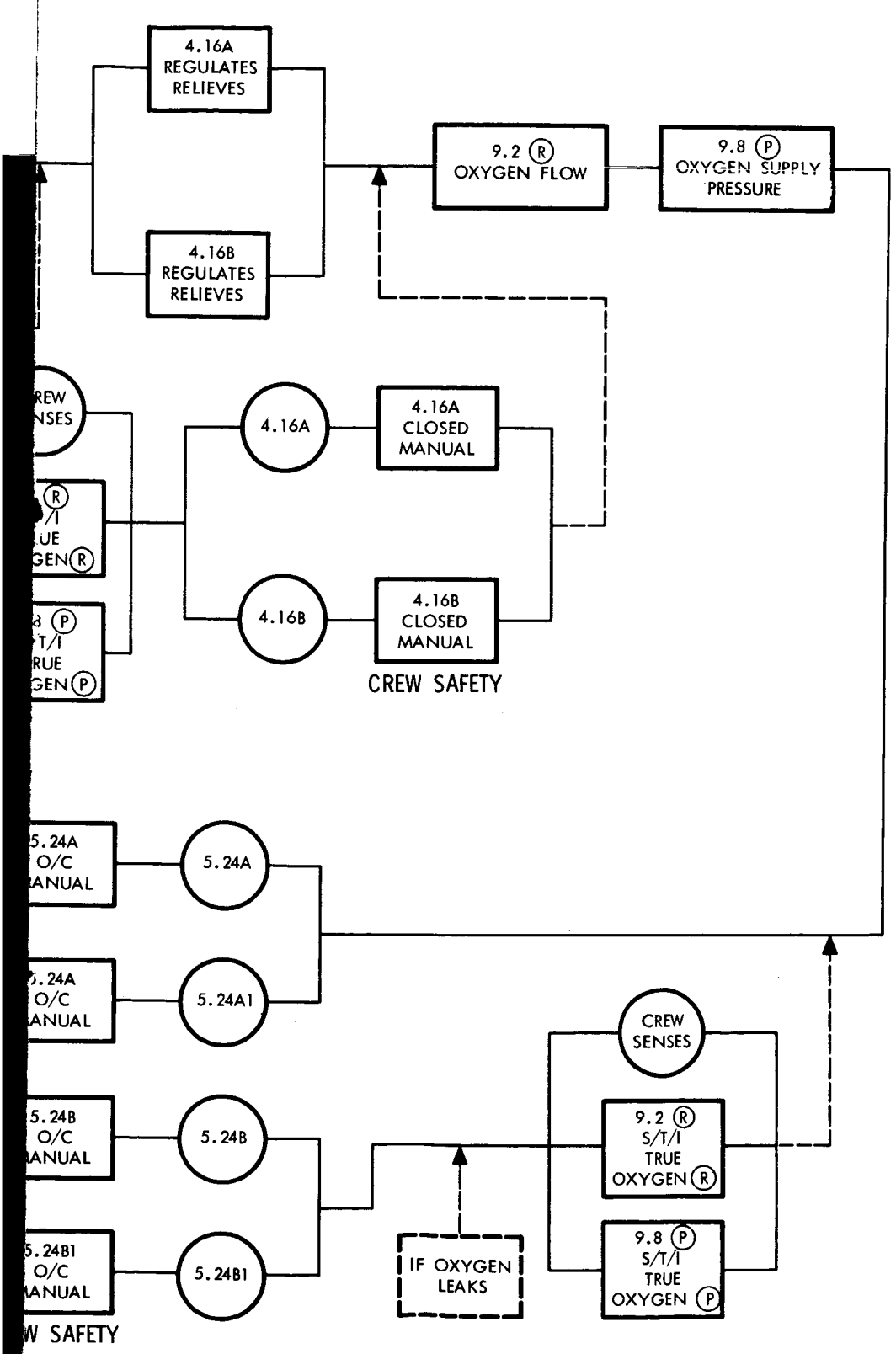
Figure 19. Command Module Pressure and Temperature Control, Logic Diagram



REENTRY AND RECOVERY MODES


4.24B
REGULATES
RELIEVES
4.24C
O/C
4.24B1
REGULATES
RELIEVES
4.24C1
O/C
4.23E
RELIEVES
4.23F
O/C
4.23E1
RELIEVES
4.23F1
O/C
4.22A
REGULATES
4.22B
REGULATES
CREW SAFETY
4.22A
4.22A
CLOSED
MANUAL
4.22B
4.22B
CLOSED
MANUAL
3.28A
REGULATES
3.28B
REGULATES
4.17
4.17
OPEN
MANUAL
3.28A
3.28A
CLOSED
MANUAL
3.28B
3.28B
CLOSED
MANUAL
CREW SAFETY
10.5 P
S/T/I
TRUE P
IF P
TOO HIGH
SUCCESS
1.29E2
O/C
MANUAL
1.29 E1.
O/C
MANUAL
CREW
SENSES
9.2 R
S/T/I
TRUE
OXYGEN R
9.8 P
S/T/I
TRUE
OXYGEN P
IF OXYGEN
LEAKS
5.24B
RELIEVES
REGULATES
5.24B1
RELIEVES
REGULATES

2



COMPONENT NUMBER	ITEM DESCRIPTION
1.29	HEAT EXCHANGER PACKAGE REGENERATIVE HEAT EXCHANGER REGENERATIVE HEAT EXCHANGER BYPASS VALVE SUIT EVAPORATOR GLYCOL-TO-SUIT AIR HEAT EXCHANGER WATER SEPARATOR WATER SEPARATOR PUMP ASSEMBLY
3.28	CABIN PRESSURE REGULATOR
4.16	DEMAND PRESSURE REGULATOR AND RELIEF VALVE
4.17	MANUAL OXYGEN SUPPLY VALVE
4.22	EMERGENCY OXYGEN INFLOW CONTROL VALVE
4.23	REENTRY OXYGEN SUPPLY SYSTEM OXYGEN TANK (2) OXYGEN FILL PORT (2) VALVE ASSEMBLY (2) PRESSURE REGULATOR (2) RELIEF VALVE (2) CHECK VALVE (2)
4.24	OXYGEN PRESSURE REGULATOR ASSEMBLY SHUTOFF VALVE (2) PRESSURE REGULATOR (2) RELIEF VALVE (2) CHECK VALVE (2)
5.24	TANK PRESSURE CONTROL AND RELIEF VALVE
9.2	SENSOR, OXYGEN FLOW RATE
9.8	SENSOR, OXYGEN PRESSURE
10.5	SENSOR, COMMAND MODULE TOTAL PRESSURE

LEGEND

○ = HUMAN OPERATION

□ = MECHANICAL OPERATION

(P) = PRESSURE

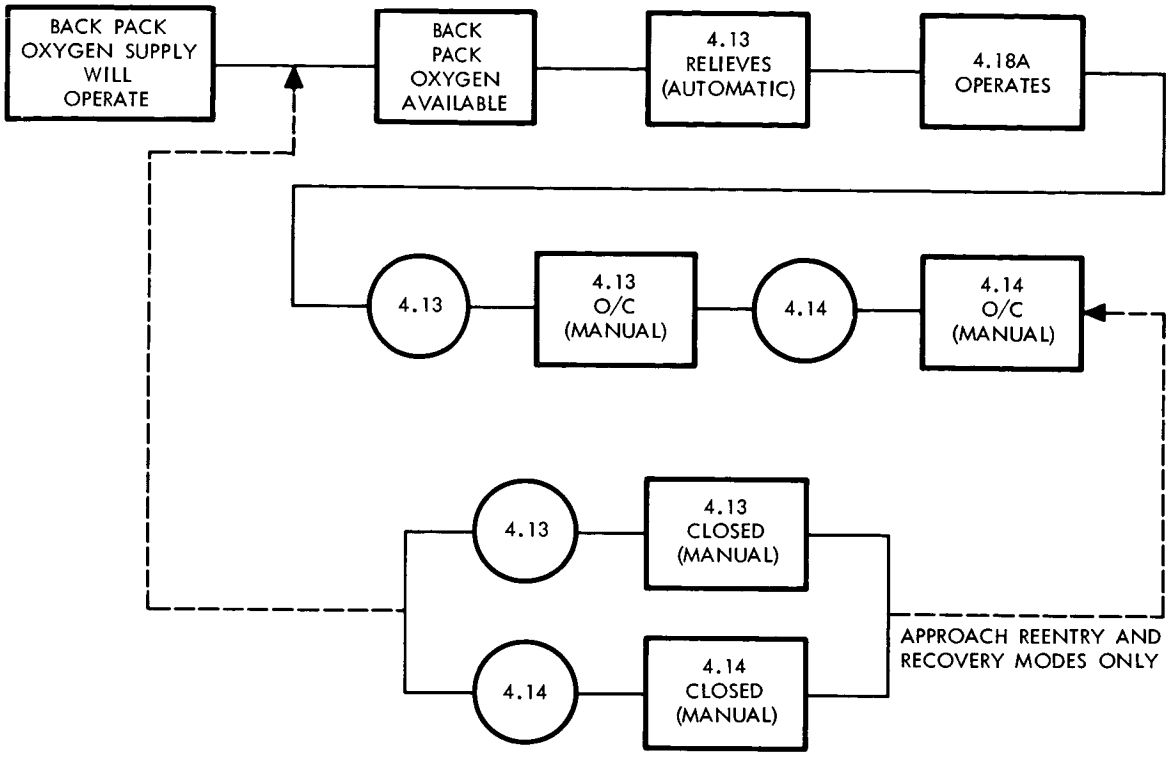
(R) = FLOW RATE

S/T/I = SENSES, TRANSMITS, AND INDICATES

O/C = OPEN OR CLOSED

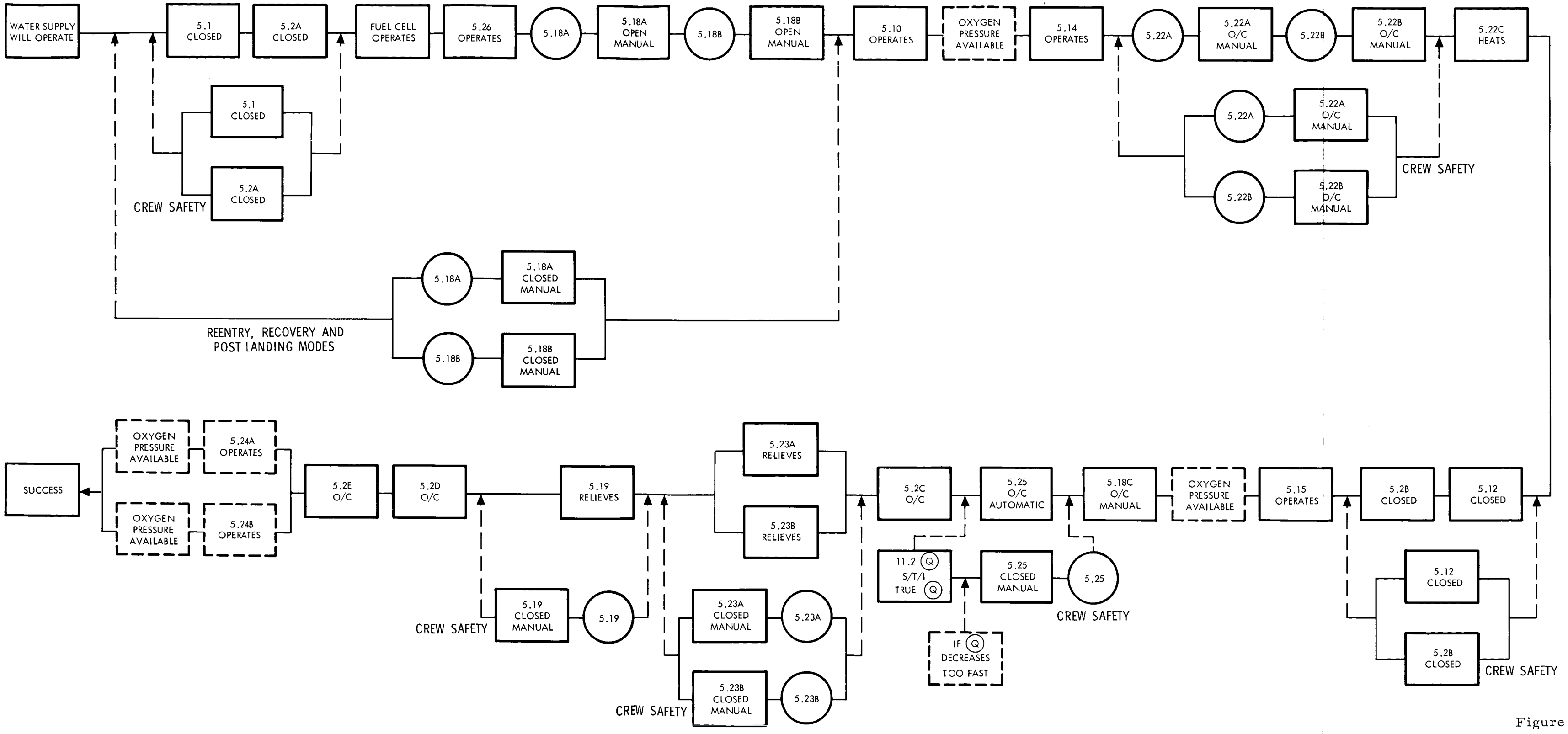
NOTE:
A, B, C, ETC., AFTER COMPONENT NUMBER INDICATES POSITION IN CIRCUIT OF SUCCESSIVE IDENTICAL COMPONENT

Figure 20. Oxygen Regulation Subsystem, Logic Diagram



COMPONENT NUMBER	ITEM DESCRIPTION
4.13 4.14 4.18	VALVE, RELIEF AND MANUAL SHUTOFF, NORMALLY CLOSED CAP, SEALING DISCONNECT, (ONE CONNECTION)
LEGEND ○ HUMAN OPERATION □ MECHANICAL OPERATION O/C OPEN OR CLOSED	

Figure 21. Back Pack Oxygen Supply, Logic Diagram



COMPONENT NUMBER	ITEM DESCRIPTION
5.1	DISCONNECT, SELF-SEALING, NORMALLY CLOSED
5.2	VALVE, CHECK, PRESET, NORMALLY CLOSED
5.10	WATER TANK, BLADDER
5.12	FREON QUICK-DISCONNECT VALVE
5.14	HEAT EXCHANGER, GLYCOL-WATER PARALLEL FLOW
5.15	WATER TANK, BLADDER
5.18	WATER SHUTOFF VALVE
5.19	WATER TANK PRESSURE RELIEF VALVE
5.22	POTABLE WATER SUPPLY ASSEMBLY
	SUPPLY VALVE (2)
	HEATER
5.23	WATER PRESSURE RELIEF VALVE
5.24	TANK PRESSURE CONTROL AND RELIEF VALVE
5.25	FREON FLOW VALVE
11.2	SENSOR, WATER QUANTITY

○ = HUMAN OPERATION
□ = MECHANICAL OPERATION
S/T/I = SENSES, TRANSMITS, AND INDICATES
Q = QUANTITY

NOTE:
A, B, C, ETC., AFTER COMPONENT NUMBER INDICATES POSITION IN CIRCUIT OF SUCCESSIVE IDENTICAL COMPONENT

Figure 22. Water Regulation, Logic Diagram

are mounted in the service module. Each radiator is divided into independent circuits. The heat rejection rate may thus be varied for different heat loads by activating more or fewer numbers of the radiator tube passages, which, in turn, vary the amount of active radiating surface. The independent circuits in each radiator also provide crew safety by incorporating the capability of eliminating a leak in a radiator section caused by meteoroid impacts. Mission success and crew safety reliability for the radiators have been apportioned at 0.99995 and 0.999998 respectively.

7. Cold plate circuit – The cold-plate circuit is also connected to the ECS water-glycol circuit. Cold plates provide the means of cooling the electronic modules in the spacecraft. Approximately forty cold plates are provided as heat exchangers. The cold plates are in contact with the electronic modules requiring cooling. Cool water-glycol circulates through internal passages in the cold plates. The cool cold plate surfaces absorb heat from the electronic modules, which is in turn transmitted to the water-glycol circulating in the cold plates.

   Mission success and crew safety reliability for the cold-plate circuit have been apportioned at 0.99995 and 0.999999, respectively.

   The Design and Thermodynamics personnel presented 33 different cold-plate circuits with varying amounts of redundancy and selectivity for cooling electronic modules. It was requested that these circuits be analyzed for the optimization of weight versus reliability.

   Approximately 50 percent of the circuits could be eliminated immediately because insufficient cold plates were included to cool the number of modules requires. After preliminary analysis, additional systems were eliminated because of weight or complexity factors.

   Further analysis of the remaining circuits resulted in the following conclusions. To achieve the high reliability values indicated for this circuit, cold plate circuits require a minimum amount of valving of the simple manually controlled on-off type. At least two individual, isolated cooling circuits should be provided, and each system should be capable of safely returning the Apollo to earth by itself. Redundant pass cold plates should be provided only for mission essential or crew safety modules.

8. Waste management system – The waste management system's purpose is to provide control of human waste. The system's function may be broken down to provide the following functions.

   Control of urine by collection, chemical treatment to kill bacteria, storage, and finally dumping into space

   Control of feces by collection and sanitized storage

   Control of debris by collection with a vacuum cleaner

Mission success and crew safety reliability for the waste management system have been apportioned at 0.999965 and 0.999998 respectively.

Three variations of the waste management system were presented by the design group for reliability analysis. The major difference in each variation is the manner in which the mode selector valve and urine disposal lock are isolated to prevent inadvertent venting of the command module atmosphere to space through these components.

In system I, the urine lock is isolated both upstream and downstream by the mode selector valve, which is, in turn, isolated and backed up by a simple open-close manually operated valve downstream in the urine overboard vent line. This system conforms to the apportioned reliability and is being developed for use in Apollo.

System II isolates the urine lock with an upstream solenoid valve and the mode selector valve downstream. The mode selector valve is in turn backed up by manual on-off valve downstream in the urine vent line.

System III employs a solenoid valve upstream from the urine lock, a solenoid valve downstream to prevent liquid (urine) flow, and a mode selector valve to prevent gas flow. The upstream solenoid valve and mode selector valve are backed up for atmospheric venting by the downstream solenoid valve.

System I, uncomplicated by additional components, that must operate sequentially, and by the relatively high failure rates of these components (solenoid valves), exhibited the high apportioned reliability presented in the preceding discussion.

Design Review

The first formal reliability design review on Apollo systems was conducted on the ECS design concept.

The Design Review Board approved the ECS design concept with the following major recommended action items.

1. Weight and balance effect studies be continued on the effects of including an additional redundant suit compressor, water-glycol pumps, and cabin fan to improve predicted mission success and crew safety probabilities

2. Filling procedures be investigated for the liquid systems (water management, water-glycol, and ground and launch freon cooling) to assure total fill and elimination of possible air entrapment

3. Further review of possible atmospheric contamination sources, control of contamination, and the possible need for in-flight contamination monitoring

4. Human factor studies on the human operations required for crew safety and possible required system changes as a result of such studies

The detailed minutes of the Design Review Board meeting are presented in Figure 23 at the end of this section.

Failure Effects Analysis

The results of the failure effects analysis performed on the ECS schematic "E" change are presented at the end of this section in Table 23.

TEST PROGRAM

The principal effort during this reporting period has been the reorientation of the environmental control subsystem qualification program. This redirection is intended to optimize the amount of engineering information that can be derived from each test article and to determine the number of test articles for qualification on the basis of equipment criticality (defined in general terms as the effect of a particular equipment item failure on crew safety). As a result of this program reorientation, the number of systems previously designated as qualification-reliability hardware has been reduced from 15 to 6. This figure includes three refurbished development systems.

Component development tests are currently being conducted on several types of check valves for use in the pressure suit circuit and water-glycol circuits. Informal reports indicate the AiResearch is experiencing some trouble in this area. The exact cause or effect will not be known until AiResearch failure analysis is received and evaluated. A performance

demonstration of the following components was witnessed at AiResearch: (1) wick-type water separators and pump assemblies, (2) centrifugal water separator, and (3) a heat exchanger package.

The demonstration of item 1 was conducted under ambient temperature and standard atmospheric pressure. The results indicated that water removal and pumping action were satisfactory under the stated conditions. This is no assurance that the same results will be achieved under mission environments.

Item 2, the centrifugal water separator demonstration, was only partially successful and indicated considerable more research and development will be required.

Item 3, the heat exchanger package, was mounted in the altitude chamber with simulated mission cabin pressure. This stage of development requires frequent adjustment and balancing of controls and flow regulations, and the results observed at this time are not a true indication of the subsystem's performance capability. Considerably more testing will be done in this area.

SUBCONTRACTOR MANAGEMENT

AiResearch

Reliability at AiResearch on the Apollo environmental system during the fourth quarter of 1962 was limited due to redirection as a result of the cost reduction program. As a result, a complete reorientation of the qualification and reliability program within the constraints of the 1963 budget was made.

The Reliability Program Plan, SS-1003-R, has been revised in accordance with S&ID critique. Submittal of this plan is being held up pending resolution of the Program Test Plan, SS-1001-R.

1. Reliability design reviews

   a. Reliability design reviews have resulted in a redesign in the water management systems with a definite improvement in crew safety and mission success.

   b. Design reviews have resulted in a redesign of the waste water separator unit. Two methods are being investigated.

1. Accumulator type, with positive expulsions and elimination of oxygen entering the waste water tanks

2. Centrifugal type, similar to the separator used in the fuel cells

c. To date, 84 percent of the ECS and GSE prototype items have been subjected to at least one design review by the subcontractor.

2. Document surveillance

Surveillance of subcontractor and S&ID documents to the subcontractor for conformance to reliability and system requirements has continued. The following documents were received and reviewed during the last reporting period.

a. AiResearch Drawing 51175, Environmental Control System Schematic Diagram, Revision C, D, and E

b. SS-1014-R, Revision 2, Design Criteria Specification

c. AiResearch Drawing 51175 D, Environmental Control System Schematic Diagram

d. SS-1000-R, Revision 4, System Specification, Environmental Control System

e. SS-1012-R, Revision 2, End Item Test Plan

f. SS-1008-R, Revision 2, Manufacturing Plan

g. SS-84100, Pressure Distribution Test Set GSE System Specification

h. SS-1001-R, Revision 2, Test Plan

i. SS-1035-R (2), Quarterly Reliability Status Report

j. SS-1013-R (6), Monthly Progress Report

k. SS-1012-R (3), Quarterly Progress Report

l. SS-844900, System Checkout Console GSE System Specification

m. SS-1022-R, Drawing List

n. SS-1013-R (7), Monthly Progress Report

o. S&ID 62-10, Statement of Work for the Apollo Environmental Control System, (12 October 1962, Revision), Rough draft, final revision and Dec., 1962 Qualification Test Addendum

3. Coordination meetings

Monthly coordination meetings are scheduled to be held on the last Monday of each month with the locale of the meeting alternating between S&ID and AiResearch. However, as a result of the effort required by S&ID and AiResearch in adjusting the program to the requirements of the cost reduction program, the monthly coordination meetings have been temporarily suspended. In the interim, meetings are called by S&ID or AiResearch as clarification and/or resolutions are required on specific problems. Meetings attended during the last reporting period and the subjects discussed are as follows:

a. October 16 - Design Review of Century System, Inc., valves

b. October 22 - Monthly Coordination Meeting

c. November 20 - ECS "D" Revision Schematic Logic Diagrams, Failure Mode and Effects Analysis, and State-of-the-Art Failure Rates

d. December 4 - Qualification Test Plan

4. Test program

a. Program test plan (SS-1001-R)

A major revision of this test plan is being prepared by AiResearch in accordance with S&ID redirection. The test plan will specify the single and combined environments, climatic-dynamics, and mission type and will incorporate the off-limit and life tests as determined to meet the requirements of crew safety and mission success with the reduced number of test items.

b. Hardware utilization

Prior to the cost reduction and qualification-reliability reorientations and test program direction, a total of 23 equivalent systems were allocated for a development, qualification and reliability test program: 18 sets to be used for development, 7 sets to be used for qualification, and 8 sets to be used for reliability demonstration. Under the revised test program, a total of eight equivalent systems have been allocated for development and qualification, five sets for development and three sets for qualification. To accomplish the desired reliability under simulated mission environments, it was planned to use three of the development systems for Phase A, Parts 1 and 2, tests. We were advised by our supplier, AiResearch, that the development hardware would not be final configuration spacecraft-type hardware and, therefore, would be unsuitable for use in the Phase A, Parts 1 and 2, of the qualification tests.

In order to implement Phase A, Parts 1 and 2, of the qualification test plan, the hardware utilization list will have to provide the required equipment.

c. Component development test

Development tests are currently being conducted at AiResearch on several types of air and liquid check valves, accumulator and centrifugal types of water separators, fluid pumps, suit compressor, and the suit circuit evaporator and circuit loop.

All development testing is conducted under reliability surveillance by means of AiResearch universal trouble reporting systems (UTRS).

d. Qualification tests

Qualification test requirements for the environmental control system have been prepared by Apollo Reliability Test Group and transmitted to the Environmental Design Group to be incorporated in SS-1000-R.

Qualification tests are scheduled for 15 March 1963. However, this is a planned date and may be unrealistic because of design considerations.

Component Suppliers

Reliability activity with subcontractors concerned with associated ECS equipment has been devoted to preparation of reliability requirements for procurement specifications and an investigation of proprietary mechanical screw-thread-type fluid fittings suitable for use in Apollo.

1. Specifications

   Reliability requirements were prepared for the following S&ID procurement specifications.

   a. MC 284-0050, Procurement Specification, Ventilating Waste Management System, ventilating check valve

   b. MC 901-0024, Procurement Specification, Waste Management System, vacuum cleaner

   c. MC 901-0025, Procurement Specification, Waste Management System, control unit

   d. MC 901-0027, Procurement Specification, Waste Management System, bacteria control unit

   e. MC 901-0029, Procurement Specification, Waste Management System, urine disposal lock

   f. MC-901-0030, Procurement Specification, Waste Management System, blower

   g. MC 901-0068, Procurement Specification, Waste Management System, backup valve

2. Fluid fitting investigation

   Although it is planned to use welded or brazed tube connections throughout the Apollo spacecraft for high reliability, a minimum amount of threaded tubing connections will be required. Because of the stringent functional, environmental and reliability requirements, most proprietary and standard screw-type fittings are incapable of satisfactory performance. S&ID's Component Technology Group, at the request of Apollo Reliability, has therefore conducted a survey of screw-thread-type fluid fitting manufacturers to determine which fittings are most likely to meet the required Apollo parameters.

Twenty-two companies were contacted. Answers were received from eighteen. Three suppliers indicated the lack of a unique fitting, one was no longer in the fitting business, and five companies requested and received more time to prepare a submittal.

The nine proposals received to date are currently being reviewed. Preliminary indications are that only two of the fittings proposed are worthy of further investigation and testing. Final judgment will be withheld, however, until the review of the proposals is completed.

PLANNED ACTIVITIES

As the ECS design undergoes further design refinement, logic diagrams and failure mode and effects analysis will be updated to conform to the changes. With these aids, numerical reliability analyses will be continually updated to determine potential reliability growth. Recommendations to optimize the system components for reliability numerical requirements will be issued.

Commencement of reliability activity on interface problems and integration of the ECS with LEM requirements are expected during the next quarter. Quarterly reliability status summaries for the ECS including recommendations for reliability improvement will be issued regularly.

Items of major importance scheduled for the next quarter are

1. Revision to Program Test Plan SS-1001-R
2. Release of prototype environmental control system development test procedure
3. Continued development testing of the prototype glycol subsystem and various subassembly components

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APOLLO-RELIABILITY/CREW SAFETY DESIGN REVIEW BOARD REPORT PAGE 1 OF 7

SUBJECT Environmental Control System DATE

REVIEW SCOPE Evaluation of System DESIGN REVIEW NO.

Reference: 1. System Specification E.C.S.-SS-1000-R Rev. 4 20 Nov. '62

2. Design Criteria Specification E.C.S.-SS-1014-R Rev. 2 12 Oct. '62

3. Failure Mode Analysis

DESIGN UNIT Mechanical System Design DEPT. NO.

TYPE OF REVIEW		RESULTS	
[X] PRELIMINARY	[ ] MAJOR	[X] APPROVED	[ ] DISAPPROVED - RESCHEDULE
[ ] OTHER	[ ] APPLICATION	OTHER:	

DISTRIBUTION

REVIEW BOARD

Approved

B. H. Hershkowitz

Reliability/Crew Safety

Design Review Chairman

R. W. Swigart

RESPONSIBLE DESIGN REVIEW ENGINEER (DATE)

REMARKS: The first Reliability/Crew Safety Preliminary Design Review resulted in exceptionally good participation. Inputs from the Review Boards evaluation, which most directly affected the conceptual design phase of the Environmental Control System, were included in the meeting agenda. A summary of the consideration items discussed (not necessarily in the order of importance) together with the board recommendations are listed below:

NO.	Consideration Summary	ACTION BY
1	First Order Abort Failure Modes	
	An abort maneuver is required on any failure mode in which the next failure will result in catastrophic loss of the Spacecraft and Crew. As a result of this type of failure the reliability goal for mission success would be reduced. The following areas were discussed as typical of this situation:	
	(a) Two compressors versus three compressors in the suit cooling loop.	
	(b) Two water-glycol pumps versus three water glycol pumps in	

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Figure 23. Design Review Board Report (Sheet 1 of 7)

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[redacted]

APOLLO-RELIABILITY/CREW SAFETY DESIGN REVIEW BOARD REPORT
(CONTINUATION SHEET) PAGE 2 OF 7

SUBJECT: Environmental Control System DATE

NO.	Consideration Summary	ACTION BY
	the cabin and service module heat exchanger cooling loops.	
	(c) Two cabin circulation fans versus three fans.	
	(d) Failure of GSE circulation control valve 2.11 to remain open during flight (reference SS-1000-R Rev. 4 Page 53)	
	(e) Failure of one of the bladders within the (Freon and waste-water) potable water or the water-glycol storage tanks.	
	The Board recommended the following:	
	(a) The use of three compressors, three fans, and three water pumps should be reinvestigated as to the effect upon space-craft weight and center of gravity restrictions in the interest of maintaining the reliability requirements for mission success.	Bouman Hershkowitz
	NOTE: It was stated that this investigation could be included with the current evaluation of the NASA proposed centrifugal water separator.	
	(b) Effort should be made to eliminate the first order failure modes that would cause mission abort such as the 2.11 valve and the storage tank bladder failures.	Bouman
2	Radiation and Meteoroid Requirements	
	The removal of the radiation and meteoroid requirements from the S&ID environmental criteria primarily reduced basic cost of the AiResearch Test Program. It was noted that S&ID will continue to monitor the material used throughout the system to insure compatability with current knowledge of radiation levels expected.	
3	Liquid System Filling Procedures	
	Provisions for venting the various liquid systems during the filling operation was not adequately described within the specification. Failure to adequately fill the water-glycol system or subsequent leakage of GO_2 into the system through the expulsion bladder would cause the water-glycol pumps to cavitate	

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Figure 23. Design Review Board Report (Sheet 2 of 7)

APOLLO-RELIABILITY/CREW SAFETY DESIGN REVIEW BOARD REPORT
(CONTINUATION SHEET) PAGE 3 OF 7

SUBJECT: Environmental Control System DATE

NO.	Consideration Summary	ACTION BY
	resulting in loss of the cooling system. In addition it was	
	indicated that water-glycol and 100% O_2 may produce some	
	unwanted by-products.	
	The Board recommended that:	
	(a) Special provisions for insuring 100% filling capability be	Bouman
	developed and included in the Specifications.	
	(b) An evaluation of the effects of glycol versus O_2	Bouman
	by-products on either water-glycol or O_2 systems should	
	be made.	
4	Water-Glycol Filtration Requirement	
	A single water-glycol filter is provided to protect the pumps,	
	valves and orifices from malfunctions caused by contaminates.	
	The following areas were discussed regarding the filter	
	requirements:	
	(a) The amount and size of contamination expected in the	Bouman
	system as a result of GSE checkout and countdown	
	circulation.	
	(b) The possibility of exceeding the contamination capability	Bouman
	of the filter in which case a by-pass would open allowing	
	collected contaminate to pass if proper design techniques	
	were not employed.	
	The Board recommended the following action:	
	(a) The specification should include contamination particle	Bouman
	size distribution and quantity requirements.	
	(b) The by-pass provisions of the filter should be adequately	
	designed to preclude washing action of the collected	
	contaminates during periods of by-pass operation.	
	(c) A method for detecting when by-pass has occurred during	Bouman
	checkout should be provided.	
	NOTE: Higher flow rate than normal through the filter	
	from GSE supply might be used to insure that by-pass will	
	not occur immediately after lift-off at normal flow rates.	

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Figure 23. Design Review Board Report (Sheet 3 of 7)

APOLLO-RELIABILITY/CREW SAFETY DESIGN REVIEW BOARD REPORT
(CONTINUATION SHEET) PAGE 4 OF 7

SUBJECT: Environmental Control System DATE

NO.	Consideration Summary	ACTION BY
	(d) Contamination level monitoring should be provided as part of GSE.	Bouman
5	Toxicity Detection and Gas Analyzer	
	Crew reaction and performance due to the presence of various gases such as glycol, R.C. fuel, and excessive CO_2 levels over extended time periods have not been thoroughly evaluated. Present state of the art is primarily based upon commercial applications. The human tolerance of glycol, on a 40-hour week, 8-hour day basis is approximately 200 PPM; however, this may not be acceptable to the crew during a period in which continuous exposure for several days is experienced. CO_2 is controlled by replacement of Lithium Hydroxide elements into the suit loop on a scheduled basis. Malfunction or failure to replace of one of the 28 units could cause CO_2 to develop in excess of safe limits which would reduce crew performance and adversely affect their health. While excessive CO_2 and glycol is not lethal in small quantities, the presence of R.C. fuel vapors is especially dangerous during the post landing phase. The present operational concept doesn't include a gas analyzer for use throughout the mission; although, it was pointed out that NASA is currently evaluating several analyzers which may be available in the future.	
	The Board recommended that the need for a gas analyzer for leak detection of the various gases and crew warning should be thoroughly investigated and justified to NASA.	Bouman
6	Fire Hazard of 100% O_2	
	Many materials including aluminum will support combustion when exposed to a fire in a pure O_2 atmosphere. Crew safety will be jeopardized by the reduced time available for them to get into their pressurized space suits before cabin depressurization can be allowed to evacuate the O_2 and the fire. It is expected that under emergency conditions the crew can put on their suits in	

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Figure 23. Design Review Board Report (Sheet 4 of 7)

APOLLO-RELIABILITY/CREW SAFETY DESIGN REVIEW BOARD REPORT
(CONTINUATION SHEET) PAGE 5 OF 7

SUBJECT: Environmental Control System DATE

NO.	Consideration Summary	ACTION BY
	less than 5 minutes; however, it was pointed out that to date	
	S&ID tests have indicated that 5 minutes has not been adequate	
	when attempted in a simulated cabineconfinement.	
	The Board made the following recommendations to help reduce the	
	hazardous effects of this condition.	
	(a) Survey all materials to be used within the crew compartment	Bouman
	to insure that they have high ignition temperature and	
	low absorbent qualities associated with 100% O_2.	
	(b) Insure that all inspection procedures for welding of	Bouman
	tubing or structure do not call for a dye penetrant which	
	can support combustion in pure O_2 gas.	
	(c) Insure that materials selected within the cabin are lint	Bouman
	and dust free.	
	(d) Continue to investigate crew reaction times associated	Bouman
	with emergency donning of space suits.	
	NOTE: Suit design and development is a NASA	
	responsibility; however, excessive use of O_2 in the process	
	of putting the suits on will adversely affect the S&ID	
	Apollo Spacecraft design. It is expected that a design	
	which is compatible with the five-minute criteria will be	
	developed, but further attempts should also be made to find	
	a way to allow immediate emergency depressurization of the	
	cabin in the event of fire by the application of pressure	
	to a partial space suit while the helmet is being put on	
	and secured for breathing.	
7	Manual Valve Operational Complexity	
	Manual positioning of valves by the crew is necessary at various	
	phases of the mission. To insure that this is not confusing to	
	the crew, Human Engineering is assisting in the positioning of	
	valve actuation levers on the control panel to be properly	
	oriented with respect to function and direction. In addition	
	to the current program the Board recommended that:	
	(a) The O_2 suit inlet valve be plainly marked for minimum	Bouman
	breathing needs to prevent inadvertent asphyxiation	

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Figure 23. Design Review Board Report (Sheet 5 of 7)

APOLLO-RELIABILITY/CREW SAFETY DESIGN REVIEW BOARD REPORT
(CONTINUATION SHEET) PAGE 6 OF 7

SUBJECT: Environmental Control System DATE

NO.	Consideration Summary	ACTION BY
	as a result of crew induced reduction of cooling	
	GO_2 flow.	
	(b) Continue developing adequate malfunction isolation	Bouman
	techniques and crew accessibility to those controls	
	which could develop into required emergency crew action.	
8	Reliability Goals for Sub-Contractor Suppliers	
	Special emphasis was made by Board members for the need to	
	acquaint sub-contractor suppliers with the same reliability	
	goals expected of S&ID and the sub-contractor. These should be	
	pointed out in accordance with NCP-200-2 as applicable to the	
	supplier.	
	No special Board recommendations were made regarding this item.	
9	Requirements for Post Landing Cabin Conditioning	
	The extremely high loads imposed by impact on land or water	
	introduces the possibility of damage to the water-glycol, R.C.	
	fuel, O_2 entry supply systems. This damage would reduce the	
	effectiveness of the post landing conditioning system. This	
	item did not result in any action item; however, some	
	attention to the crew action required immediately following	
	the landing needs to be studied in conjunction with Item 5	
	above.	
10	Future Design Changes Requiring Review	
	Normal procedure requires that significant design changes be re-	
	reviewed at the discretion of the Design Manager and Reliability	
	Manager. Two areas which could cause rereview of portions of	
	this system are:	
	(a) The centrifugal water separator system recommended by	
	NASA for removal of water from the suit loop of the	
	Environmental Control System.	
	(b) Design changes as a result of current analysis progressing	
	on the waste water versus Freon systems for the water-	
	glycol cooling circuit could cause rereview.	

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Figure 23. Design Review Board Report (Sheet 6 of 7)

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APOLLO-RELIABILITY/CREW SAFETY DESIGN REVIEW BOARD REPORT
(CONTINUATION SHEET) PAGE 7 OF 7

SUBJECT: Environmental Control System DATE

NO.	Consideration Summary	ACTION BY
11	Re-entry GO_2 Supply Purity Requirements	
	The above item was added to the agenda during the meeting to	
	point out to the Board that current purity requirements of	
	99.95% pure for the GO_2 system imposes a condition in which	
	cleaning fluids are not currently available which will satisfy	
	the requirements.	
	The Board recommended that the purity requirements of the	Morris
	re-entry GO_2 storage system be re-evaluated in view of the	
	current stage of the art cleaning fluids. It is suggested	
	that both areas be thoroughly investigated until an acceptable	
	compromise can be obtained.	

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Figure 23. Design Review Board Report (Sheet 7 of 7)

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Table 23. Environmental Control Subsystem Failure Effects Analysis

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
1.3 Valve suit circuit return	850022-1-1		Cabin gas-to-suit circuit	Fail open (cabin pressure suit pressure)	All	None	(A) (C)* None	None	None
				Fail open (cabin pressure suit pressure	All	None, in series with another check valve	(B)* None	None	Repair if failure is detected
				Fail closed	All	None	(A) (B)* None (C)* Loss of O_2 makeup	None abort	None. Don suits and connect suit circuit.
1.8 Trap, debris	811100-1-1		Remove solid contaminates from suit circuit	Main passage blocked	All	None, bypass will function	(A) (B) (C)* No debris removal	None	Clean or replace element
				**Bypass plugged	All	Loss of function	(A) (B) (C)* Loss of conditioned flow	Abort if unable to clean and repair	Don back packs during repair operation.
1.10 Compressor suit	826000-1-1		Suit circuit pressurization and circulation	Inoperative	All	None	(A) (B) (C)* Loss of gas flow	Abort	Use alternate compressor
				Operating speed low	All	None	(A) (B) (C)* Reduced gas flow and pressure	Abort	Use alternate compressor
				Burst	All	Loss of unit	(A) (B) (C)* No effect unless fragments cause other system damage.	Abort	Use alternate compressor or don back packs
1.11 Valve, suit compressor check	850052		Prevents reverse flow in compressor circuits	Fail open (associated compressor operating)	All	None	(A) (B) (C)* No effect	None	None
				Fail open (associated compressor failed)	All	Gas recirculates through failed compressor	(A) (B) (C)** Loss of gas to cabin or suits	Abort	None

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
1.11 Valve, suit compressor check (Cont)				Fail closed	All	None	(A) (B) (C)** Loss associated compressor may overheat, also loss of gas to cabin or suits	Abort	Use alternate compressor.
Part of 1.15 valve, CO_2 and odor absorber check	811400-1-1		Prevents reverse flow during cannister change.	Fail open	All	None	(A) (B) (C)* No effect (B)** Loss of gas to suits if a previous leak failure occurred to absorber	None. Abort if leakage cannot be repaired	None. Don back packs and repair.
				Fail closed	All	None	(A) (B) (C) Loss of CO_2 and odor removal capacity	None. Redundant function	More frequent cannister replacement in alternate system.
1.15 Absorber CO_2 and odor	811400-1-1		Remove CO_2 and odors from suit circuit gas.	Leakage to ambient	All	Loss of redundant absorber	(A) (B) (C)* No effect	None	Use alternate absorber, repair leak replace cannister.
					All		(B)** Loss of gas to suits if a fail open condition occurred to valve 1.13	Abort if leakage condition is not repairable	Don back packs. Replace cannister.
				Failure to remove CO_2	All	Increased CO_2 level in suit circuit	(A) (B) (C)* None redundant absorber	None	Replace cannister.
Part of 1.15 CO_2 and odor absorber package valve, CO_2 and odor absorber isolation	811400-1-1		Isolates a cannister for element replacement and non-use	Jamming in any position	All	None	(A) (B) (C) No effect (B)** Loss of gas to suits if a previous leakage failure occurred to absorber	None. Abort if leakage condition cannot be repaired	None, don back pack.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
1.16 Valve, suit bypass control	810500-1-1		Bypass excess flow from suit circuit	Fail open	All	None	(A) (B) (C)* Reduction in suit flow	None	Use manual override
				Fail closed	All	None	(A) (B) (C)* Crew comfort slightly effected	None	Use manual override.
1.17 Switch suit compressor selector	820210-1-1		Selects compressor for desired operation	Inoperative	All	None, redundant switch provided	(A) (B) (C) None redundant switch provided	None	Use alternate switch
1.23 Sensor, suit air inlet temperature	820100-1-1		Provides temperature signal for auto suit air control	Inoperative or erratic	All	None	(A) (B) (C)* Loss of automatic temperature regulation.	None	Use manual control
1.24 Control, suit air temperature	820010-1-1		Amplifies signal from temperature sensor to electrically actuated bypass valve	(A) (B) (C) Control totally inoperative or operates erratically	All	Loss of automatic temperature control for suit circuit gas.	None	None	Manually operate heat exchanger bypass valve.
1.25 Selector, suit air temperature	820020-2-1		Provides selection of desired suit circuit 925 temperature.	(A) (B) (C)* Selector totally inoperative or operates erratically.	All	Loss of automatic suit circuit 925 temperature control.	None	None	Manually operate heat exchanger bypass valve.
1.27 Sensor, suit evaporator temperature	820050-H		Provides signal to controller for supplementary cooling by water boiling in suit heat exchanger.	Sensor open or short circuited.	All	If water valve 5.5 opened, loss of H_2O if 5.5 closed, loss of cooling capacity and H_2O removal.	None if detected	Abort if excessive H_2O is lost prior to detection.	Manually operate water valve 5.5.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
1.28 Control suit evaporator air temperature	820010-1-1		Amplifies signal from 1.27 sensor to electrically actuated water valve.	(A) (B) (C)* Control totally inoperative or operates erratically.	All	For a totally inoperative control, failure effects are the same as for the 1.27 sensor above. For erratic operation, water valve 5.5 will oscillate resulting in premature wearout loss of cooling capacity and H_2O removal.	None if detected	Abort if excessive H_2O is lost prior to detection.	Manually operate water valve 5.5.
Exchanger regenerate heat, part of 1.29 heat exchanger package	813000-1-1		Cool suit circuit gas	Leakage - hot to cold	All	Reduced effect.	(A) (B) (C)* Reduction in cooling capacity and water removal	No effect unless leakage causes loss of tolerable temperature conditions, then abort.	None
				Leakage hot to ambient (cabin)	All	Drastically reduced in effect	(A) (C)* Reduction in cooling capacity and water removal increased O_2 demand. (B)* Loss of gas to suits	Abort	None
								Abort	Don back packs; repair leak if possible.
				Leakage - cold to ambient (cabin)	All	Drastically reduced effect	(A) (C)* Reduction in cooling capacity and water removal, increased O_2 demand. (B)* Loss of gas to suits	Abort	None
								Abort	Don back packs; repair leak if possible.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Valve, regenerate heat exchanger by-pass. Part of 1.29 heat exchanger package			Mix warm and cool suit circuit gas to obtain desired temperature.	Electric actuator fails or valve fails in by-pass open or closed position	All	None	None	None	None, Use manual over-ride provided on valve
Evaporator, suit. Part of 1.29 heat exchanger package			Provide supplement cooling of suit circuit gas by boiling H_2O	Leakage of suit gas to vacuum or cabin ambient	All	Loss of conditioned suit circuit gas	Depletion of O_2 supply	Abort	Don back packs
Glycol to suit heat exchanger. Part of 1.29 heat exchanger package				Leakage of glycol to suit gas or to ambient	All	Loss of cooling capacity and contamination of atmosphere	Loss of water-glycol and equipment cooling capacity	Abort	Open clycol by-pass valve
			Provides primary cooling of suit circuit gas	Leakage of suit gas to ambient	All	Loss of O_2	Depletion of O_2 supply	Abort	Don back packs
Water separator part of 1.29 heat exchanger package			Remove water from suit circuit gas	Failure to remove H_2O	All	Humidity increase to intolerable levels	Humidity increase to intolerable levels	Abort	Use back packs if humidity too high
				Leakage to ambient	All	Loss of O_2	Depletion of O_2 supply	Abort	Use back packs
Water separator pump assembly part of heat exchanger package			Pump water from separator to waste water tank	Failure of pump to operate, H_2O leak to ambient, or O_2 leak to ambient	All	Increase in humidity and/or loss of O_2 supply	None	Abort if two systems fail	Use alternate system

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Valve, suit flow shutoff and limiting part of 1:30 suit connector package and 1:31 suit flow limiter	810100-1-1		Gas pressure to cabin and suits	Fail open	All	None None	(A) (C)* None (B)* Loss of gas to suit	None No effect; abort if two or more valves fail	Repair if detectable.
				Fail closed	All	None None	(A) (B)* Loss of gass to suits (C)* No effect	No effect; abort if two or more valves fail - None	Connect suit hose to alternate valve. None
Connector suit flow return part of 1:30 suit connector package	810100-1-1		Gas from suit to suit circuit	Mechanical damage	All	Connector useless	(A) (B) (C) * None	None	Use alternate connector.
Connection suit flow supply part of 1:30 suit connector package	810100-1-1		Suit pressure and flow	Mechanical damage	All	Connection useless for suit supply	(A) (B) (C)* None	None	Use alternate connector.
Valve, suit manifold return check, part of 1.30 suit connector package	810100-1-1		Suit circuit manifold return check	Fail open	All	None	(A) (C)* None (B)* Loss of gas to suits	None None unless two valves fail	None, Connect suit hose to failed open connection.
				Fail closed	All	Loss of function	(A) (B)* Loss of gas flow to suit (C)* None	None unless two valves fail. None	Connect suit hose to alternate connection. None
2.1 Valve, space radiator outlet check	827030-1-1		Prevents reverse coolant flow.	Fail open	Space flight	None	No effect unless previous leakage has occurred in radiator.	Abort	None
				Fail closed	Space flight	Loss of heat rejection capacity	Insufficient cooling for lunar orbit	Abort	Reorient S/C for minimum heat load.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
2.2 Valve, glycol pressure relief	827530-1-1		Protects against over-pressurization in glycol circuit.	Fail closed	Space flight	None, redundant valve and shutoff provided.	Possible pressure buildup and rupture of radiator	Abort	Use alternate valve.
				Fail open	Space flight	None, redundant valve and shutoff provided.	Loss of heat rejection	None	Use alternate valve.
2.4 Valve, glycol check (valve down-stream of cabin heat exchanger)	827030-1-1		Prevents reverse coolant flow.	Fail open	All	None	Loss of heating capacity	Abort, if not able to control manually.	Manually operate valve 2.13.
				Fail closed	All	Loss of cooling capacity and/or heating capacity	During maximum heating or cooling effectiveness is reduced.	Abort, if not able to control manually.	Manually operate valve 2.13.
Valve upstream of cabin heat exchanger				Fail open	All	Loss of cooling capacity	During maximum cooling, effectiveness is reduced.	Abort, if not able to control manually	Manually operate valve 2.13.
				Fail closed	All	Loss of heating capacity	During maximum heating demand, system flow is blocked.	Abort, if not able to control manually.	Manually operate valve 2.13.
2.6 Evaporator glycol	812000-1-1		Cools glycol by water evaporation.	Leakage-glycol to steam	Space flight	None	Loss of coolant	Abort	None
				Leakage-glycol to cabin ambient	Space flight	None	Loss of coolant	Abort	None
				Leakage - cabin ambient to vacuum	Space flight	None	Loss of Oxygen	Abort	Use back packs.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
2.11 Valve, glycol shutoff	827570-1-1		Shuts off flow to radiators	Fail open	All	None	Redundant backup valve provided	None	None
				Fail closed	All	None	Loss of cooling capacity	Abort	None
2.13 Valve, cabin temperature control	812100-1-1		Shuts off flow to cabin heat exchanger	Jamming in any position	All	None, manual over ride provided	Loss of cabin temperature control	Abort, if not able to control temperature manually.	Use suits.
2.16 Switch, glycol pump selector	820210-1-1		Selects operation of glycol pumps	Inoperative	All	None	None, redundant switch provided	Abort	Use, alternate switch.
2.17 Valve, glycol shut off	827580-1-1		Allows space radiators to be bypassed by water-glycol	Fail open	Space flight	Diversion of coolant flow to radiator	Loss of cooling capacity	Abort if heat load too high.	Switch off unnecessary heat generating equipment.
				Fail closed	Space flight	None	Redundant backup valve provided	None	Use redundant valve.
2.20 Valve, space radiator isolation and vent	820220-1-1		Controls flow to sections of space radiator	Fail open	Space flight	None	None	None	None
				Fail closed	Space flight	Loss of one section of radiator	Loss of cooling capacity	Abort if heat load too high.	Switch off unnecessary heat generating equipment.
				Vent section fails open	Space flight	Loss of glycol	Loss of cooling capacity	Abort	Isolate radiator system.
				Vent section fails closed	Space flight	Unable to drain radiator when not in use	Possibility of frozen glycol mixture and rupture	Abort	Use remaining sections of radiator.
2.22 Control, glycol temperature	820010-1-1		Automatic control of glycol temperature	Inoperative or erratic operation	All	None, manually operated valve	Loss of auto glycol temperature control	None	Use manual control.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
2.23 Sensor, glycol temperature	820080-1-1		Glycol temperature sensing for automatic	Inoperative or erratic operation	All	None	None, alternate sensor provided	None	Switch to alternate sensor.
2.24 Connection, glycol fill and vent	827060-1-1		Ground servicing of glycol system	Fail open	Ground servicing	None	None, backup valves provided possible ground repair.	None	None
				Fail closed	Ground servicing	None	None detectable on ground during servicing.	None	None
2.26 Disconnect, glycol quick	827590-1-1		Prevents glycol loss, after C/M, S/M separation.	Fail open	Space flight	None	None, backup valve 2.27 provided.	None	Close valve 2.27
2.27 Valve, manual shutoff	827600-1-1		Glycol system shutoff to S/M	Fail closed	Space flight	None	Loss of heat rejection by radiator	Abort	Use suits.
				Fail open	Space flight	None	No manual shutoff for glycol	None, backup by 2.26 at S/M, C/M separation	None
2.28 Valve, glycol shutoff	827050-1-1		Backup valves in glycol system	Fail closed	All	Loss of cooling capacity (cabin)	Loss of cabin cooling or heating	Abort	Use suits.
				Fail open	All	None	None	None	None
2.29 Reservoir-glycol	812800-1-1		Provides for glycol makeup.	Bag rupture	All	Loss of function	No glycol make up available	Abort	Use suits.
				Glycol leakage	All	Loss of glycol	No glycol make up available	Abort	Use suits.
2.30 Pump assembly, glycol	850010-1-1				All				

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Accumulator			Provides makeup glycol	Leakage	All		Loss of glycol for make-up	Abort	Use reservoir glycol.
				Spring fails	All		Loss of glycol for make-up	Abort	Use reservoir glycol.
Isolation valve			Closes accumulator outlet	Fail open	Space flight	None	None	None	None
				Fail close	Space flight	Loss of function	Loss of glycol for make-up	Abort	Use reservoir glycol.
Pump				Inoperative or erratic operation	All	Loss of glycol flow and pressure	None, use alternate pump	Abort	Use alternate pump.
Valve, check			Prevents reverse glycol flow	Fail open	All	None	None, unless associated pump fails	Abort, if associated pump fails.	None
				Fail closed	All	Associated pump cannot be used.	Loss of an alternate pump	Abort	None
Filter			Filters glycol prior to entering pumps	Filter clogs	All	Reduced flow, high pressure drop	Loss of cooling capacity	Abort	None
2.31 Valve, glycol temperature control	827390-1-1		Controls glycol flow for auto temperature	Fail open	All	None	Loss of temperature controlled glycol to exchanger	None	Use manual override.
				Fail closed	All	None	Loss of temperature controlled glycol to exchanger	None	Use manual override or alternate valve.
2.32 Valve, glycol pump diverter	827610-1-1		Manual flow control valve	Fail open	All	None	Loss of glycol system circulation	None	None
				Fail closed	All	None	None	None	None

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
3.1 Valve, cabin outflow pressure regulator and negative relief	810400-1-1		Regulates pressure differential between cabin and exterior ambient	Fail open	All	Temporary drop in cabin pressure	None	None	Use manual override
				Fail closed	All	Alternate valve operates	None	None	None required unless both valves fail, then use manual override.
3.2 Exchanger, cabin heat	812100-1-1		Temperature conditions cabin gas	Glycol leak to ambient	All	Loss of conditioned gas	Loss of coolant and contamination of atmosphere	Abort	Override 2:13 temperature control valves to by pass heat exchanger and don suits if contamination severe. Use back packs and vent cabin prior to use of suits.
3.5 Selector, cabin temp	820120-1-1		Provides capability to select desired cabin temperature	Inoperative or erratic operation	All	Loss of automatic cabin temperature regulation	None	None	Manually operate 2.13 glycol temperature control valves
3.6 Anticipator, cabin temperature	820110-1-1		Provides signal of heat exchanger outlet gas temperature to 3.7 temperature control to eliminate hunting of 2.13 glycol control valves	Inoperative or erratic operation	All	Loss of automatic cabin temperature regulation	None	None	Manually operate 2.13 glycol temperature control valves
3.7 Control, cabin temperature	820010-1-1		Amplifies differential signal from 3.8 temperature sensor and 3.7 temperature anticipator to 2.13 glycol control valves	Inoperative or erratic operation	All	Loss of automatic cabin temperature regulation	None	None	Manually operate 2.13 glycol temperature control valves
3.8 Sensor, cabin temperature	820100-1-1		Senses cabin temperature and transmits signal to 3.7 temperature controller	Inoperative or erratic operation	All	Loss of automatic cabin temperature regulation	None	None	Use manual override on glycol control valve, 213.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
3.14 Valve, cabin recirculating blower selector	848010-1-1		Prevents back flow of gas through standby cabin blower	Jamming in any position	All	Inability to select alternate blower in the event of blower failure	If blower also fails, loss of capability to circulate cabin gas C.S. degraded	Abort	Don suits and use suit circuit.
3.17 Switch, cabin recirculating blower selector	820210-1-1		Allows selection of desired cabin circulation blower	Contact broken or fused together	All	Loss of standby switch capability	None C.S. degraded	None	Use alternate switch
3.18 Blower, cabin recirculating	826010-1-1		Circulates cabin gas	Inoperative or erratic operation	All	Loss of standby blower capability	None C.S. degraded	Abort	Use alternate blower. Don suits if both blowers fail.
3.27 Valve, cabin vent	850014-1-1		Provides ventillation during post landing phase	Leak to space failed to close	After earth landing	Loss of O_2 None	Depletion of O_2 supply None	Abort None	Don suits and use suit circuit. Open hatch
3.28 Regulator cabin pressure	810200-1-1		Provides regulated pressure within cabin	Fail open or closed or erratic operation	All	None	None	None	Use manual override to readjust or shut-off regulator and use alternate.
4.13 Valve, back pack supply shut off and pressure relief	827140-1-1		Controls oxygen supply for back pack charging	Fail open (relief or shutoff)	All	None	Loss of O_2 from storage if relief fails open	Abort	Use re-entry O_2 supply.
				Fail closed (relief)	All	Possible over-pressurization of back packs	None	None	Use valve to lower pressure to packs.
				Fail closed (shutoff)	All	Loss of ability to charge back packs	None	Abort	None

Primary Failure
Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
4.14 Cap, back pack supply	827140-1-1		Protects and seals connection for O_2 back pack supply	Jammed on	All	Loss of ability to charge back packs	None	Abort	None
				Damaged and will not secure	All	None	None	None	None
4.16 Valve, demand pressure regulator and relief	827300-1-1		Maintains constant O_2 pressure in suit circuit	Fail open (regulator or relief)	All	Low suit circuit pressure	Loss of O_2	None, alternate valve provided	Use alternate valve.
				Fail closed (regulator or relief)	All	None	None, alternate valve provided	None unless two valves fail then abort	Use alternate valve.
4.17 Valve, manual O_2 supply	827310-1-1		Manual control of O_2 supply to suit circuit (back up for 4.16)	Fail open	All	High suit circuit pressure	Loss of O_2 supply by flooding system	Abort	Override with shutoff valves in 4.24 O_2 pressure regulator assembly.
				Fail closed	All	None	None	None	None
4.18 Disconnect, oxygen quick	827810-1-1		Oxygen supply connection command module to service module	Leakage	All	None	Loss of O_2 supply	Abort	None
4.22 Valve, emergency O_2 inflow control	827550-1-1		Provides method for flooding command module if leakage from command module should occur	Fail open	All	High C/M press.	Loss of O_2 supply	Abort, if unable to shut off	Use Manual override in 4.24 O_2 pressure regulator assembly shut off valves.
				Fail close	All	None	None	Alternate valve provided	Use alternate valve.
4.23 Reentry oxygen supply system	827240-1-1		Supplies oxygen to suit during reentry		Reentry				
Tank, oxygen			Oxygen storage for reentry	Leakage		Loss of reentry oxygen	None, alternate tank provided	Crew safety degraded	Use alternate tank.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Port, O_2 fill			Ground servicing of O_2 tanks	Leakage		Loss of reentry oxygen	None, alternate tank provided	Crew safety degraded.	Use alternate tank.
Valve assembly			Oxygen tank shut off	Fail open		Loss of reentry oxygen	None alternate tank provided	Crew safety degraded.	Use alternate tank.
				Fail closed		Loss of available reentry oxygen	None, alternate tank provided	Crew safety degraded.	Use alternate tank.
Pressure regulator			Maintains regulated pressure to suits	Fail open		None	High pressure to suits	None	Use alternate tank.
				Fail closed		Loss of available reentry oxygen	None, alternate tank provided	Crew safety degraded.	Use alternate tank.
Relief valve			Protects reentry O_2 supply system from over-pressurization	Fail open		Loss of reentry oxygen to suits	None, alternate tank provided	Crew safety degraded	Use alternate tank.
				Fail closed		None	High pressure to suits possible	None	Use alternate tank.
Check valve			Prevents flow of O_2 to depleted O_2 storage tank	Fail open		None	Possible loss of O_2 to depleted tank	None	Close depleted storage tank shutoff valve.
				Fail closed		Loss of available reentry oxygen	None, alternate tank provided	Crew safety degraded.	Use alternate tank.
4.24 Oxygen pressure regulator assembly	827330-1-1		Supplies suit circuit, waterglycol circuit, and water supply circuit with regulated O_2.		All				

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Valve, shutoff			Shuts off O_2 from service module (in parallel lines)	Fail open		None	None	None	None
				Fail closed		None	Loss of alternate O_2 supply source	Abort	Use back packs if another failure occurs in supply system.
Valve, pressure regulator and relief			Regulates O_2 from service module (in parallel lines)	Fail open (regulator or relief)		Low O_2 pressure to environmental control system	Loss of O_2	Abort	Use alternate valve.
				Fail close (regulator or relief)		High O_2 pressure to environmental control system	Possible damage to down stream components	Abort	Use alternate valve.
Check valve			Prevents reverse O_2 flow in parallel supply lines and during reentry	Fail open		None	Loss of O_2 if another component fails in same line	None, if no other component in same line fails	None
				Fail closed		Loss of alternate supply of O_2	None	Abort if failure occurs in good line.	Use back packs.
5.1 Disconnect, water supply quick (self-sealing)	827520-1-1		Connection for ground servicing of water supply system	Leakage or fail open	All	None	None, system has check valve to prevent loss	None	None
5.2 Valve, water check	827360-1-1		Serves as back up for self-sealing quick disconnect to prevent loss of water	Fail open	All	None	None, loss of water would require disconnect to leak.	None	None
				Fail closed	All	None	None	None	None
			Prevents waste water from entering potable water system	Fail open	All	None	None	None	Shut valve 5.19.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
5.2 Valve, water check (Cont)				Fail closed	All	Loss of ability to replenish waste water supply from fuel cells	Reduced cooling capacity during maximum demand	Crew safety degraded.	Reduce thermal loads on environmental control system
			Serves as backup for freon self-sealing quick disconnect to prevent loss of waste water.	Fail open	All	None	None, loss of water would require disconnect to leak	None	None
				Fail closed	All	None	None	None	None
5.5 Valve, suit evaporator water inflow control	827010-1-1		Controls flow of water for evaporation to provide additional cooling of suit circuit gas	Fail open	All	Depletion of H_2O	Improper cooling at suit evaporator	None	Use manual override if failure is detected.
				Fail closed	All	Loss of cooling capacity	Loss of automatic cooling control during maximum demand	Degraded, valve has manual override	Control valve manually during maximum cooling demand.
5.10 Tank, potable water	812300-1-1		Stores metabolic and hygenic water for crew use	Bag rupture	All	Inability to expel water from tank	Loss of cooling capacity	Abort	Use emergency water supply.
				Leakage	All	Loss of water	Loss of cooling capacity	Abort	Use emergency water supply.
5.12 Disconnect, freon quick self-sealing	827520-1-1		Connection for freon to cool command module during prelaunch activities	Fail open	All	None, system has check valve as back up to prevent loss of water after launch	None	None	None
5.14 Chiller, water	812400-1-1		Cools potable water with glycol for crew use	Leakage-water to ambient	All	Loss of potable water	None	Abort	Use emergency water supply.
				Leakage-glycol to water	All	Contaminates potable water and loss of glycol	Loss of glycol cooling capacity	Abort	Use emergency water supply.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
5.15 Tank, waste water	812200-1-1		Supplies evaporators with water for extra cooling capacity during maximum demand	Bag rupture	All	Inability to expel water from tank	Loss of extra cooling capacity during maximum heat loads	Abort	None
				Water leakage	All	Loss of water	Loss of cooling capacity	Abort	None
5.18 Valve, water shut off	827410-1-1		1. Shuts off water flow to water tanks during ground servicing and during fuel cell operation	Fail open	All	None	None	None	None
				Fail closed	All	Unable to fill potable tank or waste tank during mission	Loss of extra cooling capacity and potable water aft tanks are dry	Abort	Use emergency drinking water supply.
			2. Controls flow at waste water tank	Fail open	All	None	None	None	None
				Fail closed	All	Loss of available waste water for cooling	Loss of cooling capacity	Abort	None
5.19 Valve, water tank pressure relief	827820-1-1		Priority valve allows potable water tank to be filled before waste water tank.	Fail open	All	Loss of potable water to waste water system	None, manual override on valve	None	Close valve manually.
				Fail closed	All	Possible damage to potable water tank	None, manual override on valve	None	Cpen valve manually.
5.25 Valve, freon flow	827840-1-1		Controls flow of freon flow cooling during prelaunch activities	Fail open	All	Loss of cooling water to glycol evaporator	Excessive glycol cooling	None, manual override on valve	Close valve manually.
				Fail closed		No cooling during prelaunch	Unable to cool glycol during prelaunch	None, manual override on valve	Manually operate valve.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
5.26 Disconnect water, quick	827850-1-1		Connects water supply from service module to command module and ground supply	Leakage	All	Loss of potable water from fuel cell	Loss of water from fuel cell during maximum cooling demand	Abort	None
5.22 Potable water supply assembly valve, supply	812900-1-1		Provides crew with metabolic and hygenic water		All				
			Cold water supply	Fail open		Loss of potable water	Loss of cooling capacity during emergency or maximum demand	Abort	Use emergency supply.
			Hot water supply	Fail closed		Unavailable cold drinking water	None	None	Use hot water or emergency supply.
			Hot water supply	Fail open		Loss of potable water	Loss of cooling capacity during emergency or maximum demand	Abort	Use emergency supply.
				Fail closed		Unavailable hot water	None	None	Use cold water.
Heater			Heats water for crew use	Inoperative		Unable to provide hot water	None	None	Use cold water.
5.23 Valve, water pressure relief	827830-1-1		Water evaporator systems pressure relief	Fail open	All	Loss of cooling water	None, alternate valve provided	None, unless two valves fail then abort	Use alternate valve.
				Fail closed		Possible overpressurization of evaporator systems	None, alternate valve provided	None, unless two valves fail then abort	Use alternate valve.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

Table 23. Environmental Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
5.24 Tank pressure control and relief valve	810300-1-1		Controls oxygen pressure for expelling water and gylcol from tanks	Fail open (pressure control)	All	High pressure to water and glycol systems	None, alternate valve provided	None unless two valves fail then abort	Use alternate valve.
				Fail closed (pressure control)	All	No O_2 pressure to water or glycol systems	None, alternate valve provided	None unless two valves fail then abort	Use alternate valve.
				Fail open (relief valve)		Loss of O_2	None, alternate valve provided	None unless two valves fail then abort	Use alternate valve.
				Fail closed (relief valve)		High pressure to water and glycol system	None, alternate valve provided	None unless two valves fail then abort	Use alternate valve.

* Primary Failure
** Secondary Failure
(A) In suits, cabin pressurized
(B) In suits, cabin unpressurized
(C) In shirtsleeves

COMMAND MODULE REACTION CONTROL SUBSYSTEM

During the quarter beginning 1 October 1962, primary engineering effort was directed toward a continued attempt to select a command module reaction control system design configuration that would have acceptable performance levels, including reliability. The latest design configuration under consideration (i.e., propellant system - dual) has been analyzed and is described herein. The reaction control system reliability requirement is 0.999969 and can be achieved with the component reliability values allocated. Present state-of-the-art component reliabilities require improvement since system reliability synthesized from state-of-the-art values is only 0.993473.

This report includes (1) a description of a reliability study that compares six design configurations, (2) a failure effects analysis that considers all pertinent modes of failure, and (3) a rocket engine report and test program.

CONFIGURATION ANALYSIS

Command Module RCS Description

The purpose of the reaction control system (RCS) is to maintain spacecraft attitude control in the absence of atmosphere by providing controlled thrust about the three spatial axes. The command module RCS is specifically required to provide this attitude control for the command module after separation from the service module. Reorientation of the command module must be accomplished before reentry so that the heat shield will be in the required reentry position. During reentry, positive attitude control is absolutely necessary to achieve the highly restricted reentry window, flight path envelope, and parachute deployment. The system is designed to operate without gravity requirements and can be subdivided into pressurization, reactant supply, and engine bank subsystems. The pressurization subsystem provides, at the proper time, carefully regulated gas pressure to expell the hypergolic reactants from the reactant storage and supply subsystem in the required ratio to the reaction engine bank.

Reliability Studies

A reliability analysis, comparing six proposed configurations, was completed during this quarter on the command module RCS. Table 24 presents a summary of the results:

Table 24. Results of Reliability Analysis

Configuration	State-of-the-Art Reliability	Apportioned Reliability
1 (Ref. Figure 22)*	0.99601	0.999980
2 (Ref. Figure 23)*	0.99606	0.999984
3 (Ref. Figure 24)*	0.99624	0.999978
4 (Ref. Figure 25)*	0.98560	0.999967
5 (Ref. Figure 26)*	0.99036	0.999978
6 (See Figure 24 herein)	0.99347	0.999969
Command module RCS reliability requirement: 0.999969		

* Reference SID 62-557-3, "Third Quarterly Reliability Status Report"

The following is a brief description of each configuration:

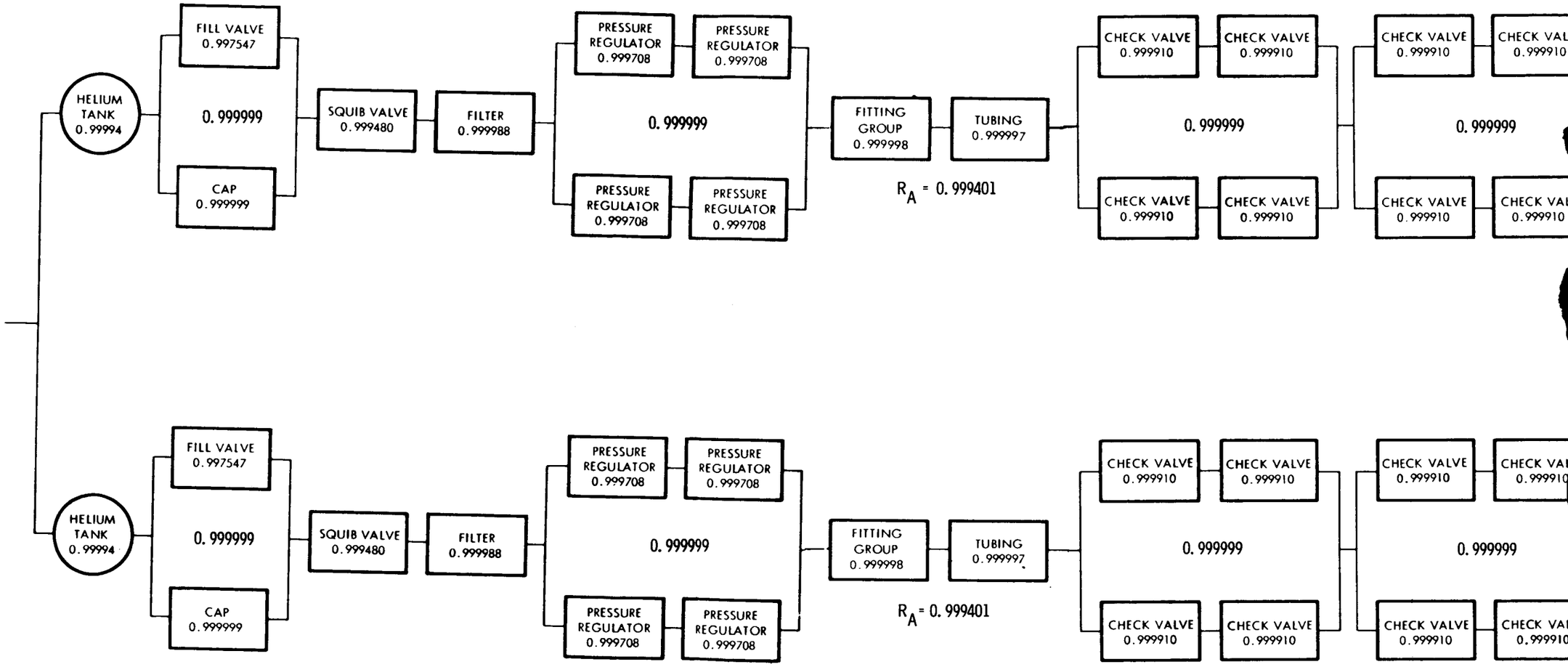
Configuration 1
(a) Dual system with propellant interconnects
(b) Parallel vapor check valves

Configuration 2
(a) Dual system with propellant interconnects
(b) No vapor check valves

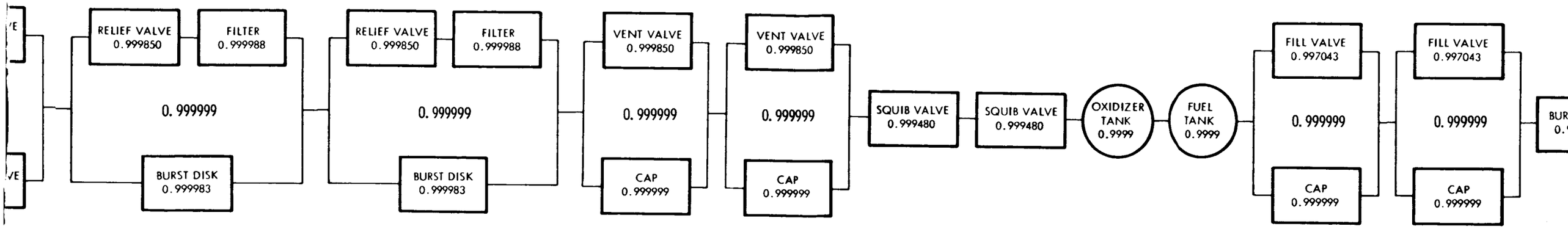
Configuration 3
(a) Integrated helium subsystem
(b) Dual propellant subsystem with interconnects
(c) Dual engine bank

Configuration 4
(a) Triple propellant subsystem with interconnects
(b) Dual helium subsystem with interconnects
(c) Dual engine bank

A. PRESSURIZATION



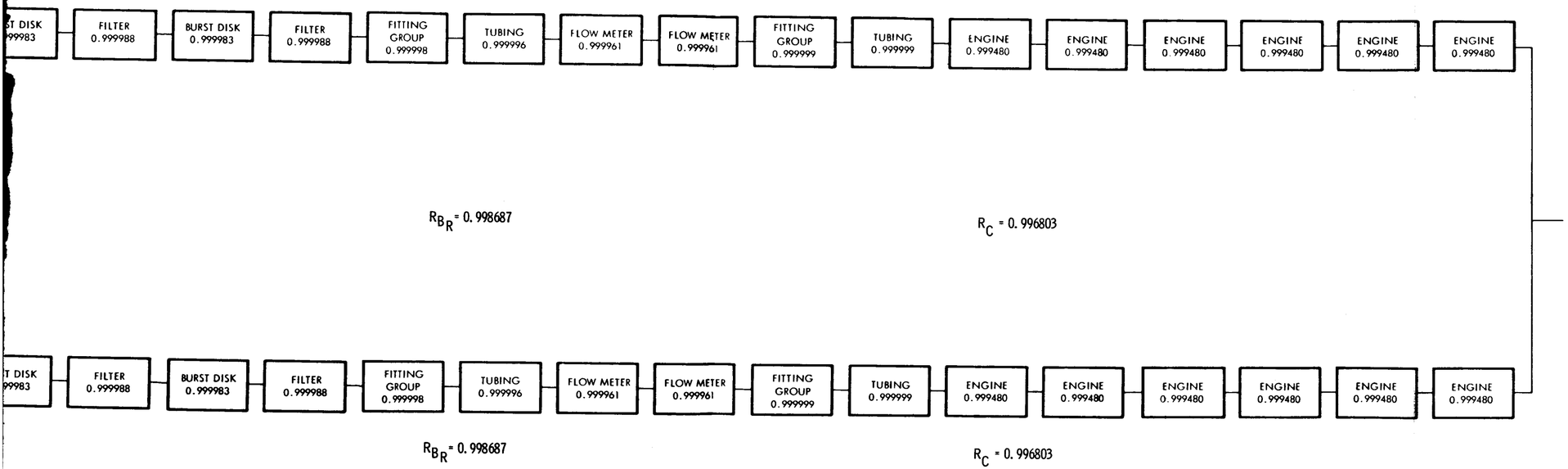
B_R REACTANT SUPPLY



RELIEF VALVE 0.999850
FILTER 0.999988
0.999999
BURST DISK 0.999983
RELIEF VALVE 0.999850
FILTER 0.999988
0.999999
BURST DISK 0.999983
VENT VALVE 0.999850
0.999999
CAP 0.999999
VENT VALVE 0.999850
0.999999
CAP 0.999999
SQUIB VALVE 0.999480
SQUIB VALVE 0.999480
OXIDIZER TANK 0.9999
FUEL TANK 0.9999
FILL VALVE 0.997043
0.999999
CAP 0.999999
FILL VALVE 0.997043
0.999999
CAP 0.999999

2

C. ENGINE PACKAGE



Mathematical Model For The Command Module Reaction Control System

Propellant System Dual Configuration Reliability = $R_{(PSD)} = R^2 + 2 R q R_{vo}$

in which : $R = R_A R_B R_C$, $q = 1 - R$

where : $R_A = R_{\text{helium tank}} (R_{\text{fill valve}} + R_{\text{cap}} - R_{\text{fill valve}} R_{\text{cap}})$
$R_{\text{squib valve}} R_{\text{filter}} (2 R_{\text{regulator}}{}^2 - R_{\text{regulator}}{}^4)$
$R_{\text{fittings}} R_{\text{tubing}}$

$R_B = (2 R_{\text{check valve}}{}^2 - R_{\text{check valve}}{}^4)^2 (R_{\text{relief valve}} R_{\text{filter}} + R_{\text{burst disk}} - R_{\text{relief valve}} R_{\text{filter}} R_{\text{burst disk}})^2 (R_{\text{vent valve}} + R_{\text{cap}} - R_{\text{vent valve}} R_{\text{cap}})^2 R_{\text{squib valve}}{}^2 R_{\text{ox tank}} R_{\text{fuel tank}} (R_{\text{fill valve}} + R_{\text{cap}} - R_{\text{fill valve}} R_{\text{cap}})^2 (R_{\text{burst disk}} R_{\text{filter}})^2 R_{\text{fittings}} R_{\text{tubing}}$

$R_C = R_{\text{flow meter}}{}^2 R_{\text{fittings}} R_{\text{tubing}} R_{\text{engines}}{}^6$

and : $R = 0.994898$, $R^2 = 0.989822$, $q = 0.005102$, $R_{vo} = 0.999522$

therefore : $R_{(PSD)} = 0.989822 + 2 (0.994898) (0.005102) (0.999522)$
$= \underline{0.999969}$

Definitions:
R_A = Reliability of Helium Pressurization Sybsystem
R_B = Reliability of Propellant Supply Subsystem
R_C = Reliability of Engine Package Subsystem
R_{vo} = Reliability of Solenoid Valve Operation (two valves)
R = Reliability of one side of the dual system
q = Probability of failure of one side of the dual system
$R_{(PSD)}$ = Reliability of Dual System

3

Figure 24. Command Module Reaction Control System Logic Diagram

Configuration 5 (a) Integrated helium subsystem
(b) Triple propellant subsystem with interconnects
(c) Dual engine tank

Configuration 6 (a) Dual system with no interconnects
(b) Quadruple regulators and check valves

Configuration 6 was selected because it requires a minimum of crew action relative to switching in alternate modes and because it meets the reliability requirement with a minimum of weight.

Table 25 presents the component Reliabilities used in this study.

Configuration 6

This configuration is of the latest design, has been tentatively adopted by propellant systems, and is a composite version of previously evaluated designs. The recommendations (by Apollo Reliability) that were adopted included (1) dual diaphram reactant expulsion tanks, (2) quadruple helium pressure regulators, and (3) quadruple helium check valves upstream of the reactant tanks. The only reliability recommendation not adopted was the reactant supply interconnect. This feature would have permitted the emergency use of the engine bank of one system with the propellant supply of the other. The recommendation was not incorporated because of its possible requirement for crew intelligence and valve operation during the severe reentry environment. Figure 24 presents the logic block diagram and mathematical models for this configuration.

Failure Effects Analysis

A command module RCS failure effects analysis has been completed. It outlines each type of failure for every item in the system and evaluates the effect of these failures on the performance of the system components, the system itself, and the spacecraft as a whole. The analysis is presented in Table 26 at the end of this section. This failure effects analysis justifies the redundancy requirements illustrated in Figure 24 that were initially determined by preliminary considerations.

Rocket Engines

Rocketdyne, manufacturer of the Apollo command module RCS engines, has held design reviews covering the thrust chamber assembly and the propellant valves. A preliminary rocket engine failure mode analysis has been performed; however, corrections are needed before the analysis is complete. A detailed reliability apportionment has been performed concurrently with this failure mode analysis.

Table 25. Command Module RCS State-of-the-Art and Apportioned Component Reliabilities

Component	Failure* Rate x 10^{-6}	Operating Time (hr or cycle)	State-of-the-Art Reliability
1. Burst disk	17**	1 cycle	0.999983
2. Connections hydraulic	29	0.75 336	0.999978 0.990256
3. Filter, mechanical (in line)	16	0.75	0.999988
4. Flow meter	52**	0.75	0.999961
5. Plug, cap (use fitting)			
6. Regulator, pressure, pneumatic	390	0.75	0.999708
7. Tank, helium (pressure)	4.2	336	0.998589*
8. Tank, positive expulsion (dual diaphragm)	83**	336	0.972112*
9. Valve, check	120	0.75	0.999910
10. Valve, fill (fuel)	8.8**	336	0.997043
11. Valve, fill (pressure)	7.3**	336	0.997547
12. Valve, relief (pressure)	200	0.75	0.999850
13. Valve, solenoid, multiple path (operational mode)	239	1 cycle	0.999761
14. Valve, solenoid (stand-by mode)	5.2**	0.75	0.999996
15. Valve, squib (parallel filament)	520**	1 cycle	0.999480
16. Valve, vent (pressure)	200	0.75	0.999850
17. Rocket engine	5000**	0.75	0.996250*

*Procurement specification reliability values that are different from this table:

7. Helium tank = 0.999940
8. Positive expulsion tank = 0.999900
17. Rocket engine = 0.999480

**Compiled or extrapolated values

A rough draft of the reliability program plan has been completed but requires revision to reflect the redirected scope of reliability testing as required by the revised procurement specification. The revised Reliability Program Plan, Rocket Engine Failure Mode Analysis, and Reliability Apportionment Report are scheduled for release during the forthcoming quarter.

TEST PROGRAM

Command Module Rocket Engine

Rocketdyne is currently writing a test program and preparing test procedures and specifications for engine components. The engine procurement specification (MC 901-0067A) was revised by S&ID as a result of qualification program redirection. Rocketdyne's schedule for the next quarter includes development testing on valves and the thrust chamber.

Reaction Control Propellant and Pressurization Subsystems

Propellant seal and 0-ring compatability tests and methods of brazing and welding tube connections are being evaluated at S&ID Engineering Development Laboratories. The fixture to be used for system evaluation of the reaction control systems is complete and testing of components for the Phase 1 RCS breadboard test program is now in progress. Development testing of components for the Apollo reaction control propellant and pressurization subsystems will commence during the next quarter

Effects of Qualification Program Reorientation

As a result of the recent qualification program redirection, a total of 68 Rocketdyne command module component RCS engines were deleted from the test program, Also, the number of tests was reduced, a total of 17 Rocketdyne engine solenoid valves were deleted, and the maximum number of components within the propellant feed system was reduced from 12 to 7. The actual number of components allowed for testing is based upon criticality. The requirements for sand, dust, and salt spray tests were also deleted.

PLANNED ACTIVITY

A combined safety and reliability study is planned that will consider command module RCS rocket engine failures, using the results of a recent thermo and fluid dynamics engine analysis

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Helium storage tank			Store pressurization gas	Rupture	All	Loss of pressurization	Render inoperative	Loss of redundancy	None
				Slow leak	1 to 58	Eventual loss of pressurization	Eventually render inoperative	Eventual loss of redundancy	None
				Support failure	All	Probably precipitate tubing failure (restriction or rupture)	Probably render inoperative or system degradation	Probable loss of redundancy, and possible structural damage to systems or spacecraft.	None
Helium fill valve with cap			Inflight: maintain high-pressure seal Ground: provide GSE access	Rupture	All	Loss of pressurization	Render inoperative	Loss of redundancy	None
				Slow leak	1 to 58	Eventual loss of pressurization	Eventually render inoperative	Eventual loss of redundancy	None
Pressurization squib valve.			Isolate high pressure until desired	Failure to operate when energized	56, 57	No pressurization	Render inoperative	Loss of redundancy	None
				External leak	1 to 58	Eventual loss of pressurization	Eventually render inoperative	Eventual loss of redundancy	None
				Internal leak	1 to 58	Premature pressurization	Exceed design operating times, possibly precipitate additional failures	None	None
				Premature operation	1 to 58	Premature Pressurization	Exceed design operating times, possibly precipitate additional failures	None	None
Helium filter			Trap foreign particles from squib or gas	Failure to trap particles	56, 57	Possible interference with operation of regulators	Possible loss of redundancy	None	Operate solenoid valve of unregulated path.

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Helium filter (Cont)				Clogged filter	56, 57	Interference with gas flow and possibly with regulation	Possible system degradation	Possible loss of redundancy	None
				External leak	56, 57	Eventual loss of pressurization	Eventually render inoperative	Loss of redundancy	None
Helium solenoid valve			Isolate unregulated path,	Failure to operate when energized	56, 57	Loss of pressurization (but precludes two regulator failures)	Render inoperative (but precludes two regulator failures)	Loss of redundancy	None
				Premature operation	1 to 58	None	None	None	Reopen
				External leak	56, 57	Eventual loss of pressurization	Eventually render inoperative	Eventual loss of redundancy	Valve cycling
				Internal leak (precludes operation)	56, 57	Eventual loss of pressurization gas via relief valve	Render inoperative	Loss of redundancy	Valve cycling
Helium pressure regulator			Reduce to operating pressure and maintain.	Improper regulation	56, 57	Loss redundancy	None	None	None
				Fail closed	56, 57	Loss of redundancy	None	None	None
				Fail open	1 to 58	Loss of redundancy	None	None	Operate isolating solenoid valve
				External leak	65, 57	Eventual loss of pressurization	Eventually render inoperative	Loss of redundancy	Operate isolating solenoid valve
Helium fitting			Provide connections and maintain pressure seal.	External leak (or rupture)	All	Loss of pressurization	Render inoperative	Loss of redundancy	None
Helium tubing			Distribute, confine, and retain pressurizing gas.	Rupture	All	Loss of pressurization	Render inoperative	Loss of redundancy	None
			Distribute, confine and retain pressurizing gas.	Restricted flow from "pinch" or excessive bend	56, 57	Reduced flow pressure	Reduced performance	Up to loss of redundancy	None

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Helium check valve			Maintain one-way flow of pressurizing gas and separation of leaking reactants.	Fail closed	56, 57	Loss of redundancy	None	None	None
				Fail open	all	Explosion (if permeation of both fuel and oxidizer diaphragm exceeds limits otherwise, none)	Explosion (if permeation of both fuel and oxidizer diaphragm exceeds limits otherwise, none) None	Loss of Apollo (if permeation of both fuel and oxidizer diaphragm exceeds limits otherwise, none) None	None
				External leak	56, 57	Eventual loss of pressurization	Eventually render inoperative	Loss of redundancy	None
Helium vent valve with cap			Inflight: maintain a pressure seal. Ground: provide GSE pressure equilization.	External leak or rupture	1 to 58	Loss of pressurization	Render inoperative	Loss of redundancy	None
Helium burst disk			Maintain pressure seal and relieve excess pressure	Burst above designed pressure	56, 57, 58	Depends upon burst pressure, if above system limit, then secondary damage is possible; otherwise, just loss of pressure	Render inoperative	Loss of redundancy	Timely actuation, and depressurization system.
				Burst below designed pressure	56, 57	Loss of pressurization	Render inoperative	Loss of redundancy	None
				Internal leak	56, 57	Loss of redundancy	None	None	None
				External leak	56, 57	Eventual loss of pressurization	Eventually render inoperative	Loss of redundancy	None
Burst disk filter			Trap foreign particles from burst disk or gas	Failure to trap particles	56, 57, 58	Possible interference with relief valve (precludes operation of burst disk)	Possibly render inoperative (precludes operation of burst disk)	Possible loss of redundancy (precludes operation of burst disk)	None

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Burst disk filter (Cont)				Clogged filter	56, 57, 58 and postlanding	Possibly exceed design pressure of system (precludes operation of burst disk)	Possibly render inoperative	Possible loss of redundancy	Timely actuation of system and depressurize system
				External leak	56, 57	Eventual loss of pressurization	Eventually render inoperative	Loss of redundancy	None
Helium relief valve			Back up to burst disk (i.e., maintain seal and relieve excess pressure)	Fail closed	56, 57, 58	Possibly exceed design pressure of system	Possibly render inoperative	Possible loss of redundancy	Timely actuation of system, and depressurize system
				Fail open	56, 57	Loss of pressurization	Render inoperative	Loss of redundancy	None
				External leak (All failures preclude operation of burst disk.)	56, 57	Eventual loss of pressurization	Eventually render inoperative	Loss of redundancy	None
Helium depressurization squib valve			Release pressurizing gas when desired	Failure to operate when energized	58	Possibly exceed design safety limits	Possible rupture of system	Possible explosion (pressure or chemical)	System actuation below excessive pressures
				Premature operation	1 to 58	Loss of pressurization	Render inoperative	Loss of redundancy	None
				External leak	1 to 58	Eventual loss of pressurization	Eventually render inoperative	Loss of redundancy	None
Positive expulsion tank			Provide reactant on demand in any attitude and from zero to maximum g loads	Rupture	all	Loss of pressure	Render inoperative	Loss of redundancy and possible corrosive action or fire	First attempt to exhaust reactant, then depressurize system.
				External leak	all	Eventual loss of pressure	Corrosive action and possible fire or explosion	Loss of redundancy. Corrosive action and possible fire or explosion	First attempt to exhaust reactant, then depressurize system.

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Positive expulsion tank (Cont)				Support failure	all	Possible tubing failure	Possibly render inoperative and possible corrosive action and fire or explosion	Possible loss of redundancy and possible corrosive action or fire or explosion	First attempt to exhaust reactant, then depressurize system.
				Failure of positive expulsion device-leak or excessive permeation (internal)	all	Reduced flow pressure; possible loss of pressure and possible explosion (precludes permeation of both tanks)	Render inoperative and possible explosion	Loss of redundancy and possible fire or explosion	First attempt to exhaust reactant, then depressurize system.
				Failure of positive expulsion device (to expel)	56, 57	No reactant flow	Render inoperative	Loss of redundancy	First attempt to exhaust reactant, then depressurize system.
Reactant fill valve with cap			In-flight: Maintain pressure seal	Rupture	all	Loss of reactant	Render inoperative	Loss of redundancy and possible corrosive action or fire	First attempt to exhaust reactant, then depressurize system.
			Ground: Provide GSE accessability	Leak	all	Eventual loss of reactant	Eventually render inoperative	Loss of redundancy and possible corrosive action or fire	First attempt to exhaust reactant, then depressurize system.
Reactant burst disk			Isolate reactant until desired	Burst above designed pressure (e.g., failure to burst.)	56, 57	No reactant flow	Render inoperative	Loss of redundancy	Ascertain that all downstream valves are open.
				Burst below designed pressure (or internal leak)	1 to 58	Exceed design operating times	None	None	None
				External leak	all		Eventually render inoperative	Loss of redundancy. Corrosive action and possible fire	First attempt to exhaust reactant, then depressurize system.

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Reactant filter			Trap foreign particles from burst disk or reactant	Failure to trap particles	56, 57	Possible interference with engine injector operation	Possible engine failure	Possible loss of redundancy	Operate isolating valves if necessary.
				Clogged filter	56, 57	Possible interference with reactant flow	System degradation	Possible loss of redundancy	None
				External leak	56, 57, 58 and postlanding	Eventual loss of reactant	Eventually render inoperative and possible corrosion action and fire	Loss of redundancy and possible corrosive action and fire	Attempt to exhaust reactant, then depressurize system.
Reactant fittings			Provide connections and maintain seal	External leak (or rupture)	all	Eventual loss of reactant	Eventually render inoperative and possible corrosive action and fire	Loss of redundancy and possible corrosive action and fire	Attempt to exhaust reactant, then depressurize system.
Reactant tubing			Distribute, confine, and retain reactant	Rupture	all	Loss of reactant	Render inoperative and possible corrosive action and fire	Loss of redundancy and possible corrosive action and fire	Attempt to exhaust reactant, then depressurize system.
				Restricted flow from pinch or bend	56, 57	System degradation	Possibly render inoperative	Possible loss of redundancy	None
Reactant solenoid valve			Isolate abnormally operating engine bank	Failure to operate when energized	56, 57, 58	Possible explosion or uncontrolled burning	Possible explosion or uncontrolled burning	Possible explosion or uncontrolled burning	Depressurize system.
				Premature operation	1 to 58	Single (or no) reactant flow - no thrust	Render inoperative, possible corrosive action and fire	Loss of redundancy, possible corrosive action and fire.	Cycle valve, and depressurize system.
				External leak	56, 57, 58 and postlanding	Eventual loss of reactant	Eventually render inoperative possible corrosive action and fire	Loss of redundancy, possible corrosive action and fire.	Cycle valve, exhaust reactant, and depressurize system.

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Reactant solenoid valve (Cont)				Internal leak (precludes engine failure and valve operation)	56, 57, 58 and postlanding	Loss of reactant, possible corrosive action and fire	Possible corrosive action and fire	Possible corrosive action and fire	Cycle valve, and depressurize system.
Reactant flow meter			Indicate to crew compartment the rate of reactant flow to engines	Fail closed	56, 57	Single reactant flow, no thrust	Render inoperative	Loss of redundancy	None
				Failure to indicate	56, 57	None	None	None (unless there are additional failures)	None
				External leak	56, 57, 58 and postlanding	Eventual loss of reactant	Eventually render inoperative, possible corrosive action and fire	Loss of redundancy, possible corrosive action and fire	Operate solenoid valves.
				Restricted flow	56, 57	System degradation, possible fire or explosion	System degradation, possible fire or explosion	Loss of redundancy	Operate solenoid valves.
Rocket engines			Provide thrust about control axes	Explosion	56, 57, 58 and postlanding	Loss of one axial thrust vector	System degradation, possible additional damage	Loss of redundancy, possible spacecraft damage or loss	Operate solenoid valves.
				Uncontrolled burning, fire	56, 57, 58 and postlanding	Loss of use	Render inoperative	Loss of redundancy, possible crew loss	Operate solenoid valves.
				Excessive ablation of throat	56, 57	Loss of use, possible fire	Render inoperative	Loss of redundancy	Operate solenoid valves.
				Failure to start	56, 57	No thrust for one (or more) axial vector	System degradation	Loss of redundancy	Cycle engine valves.
				Improper reactant ratio	56, 57	Reduced thrust, no thrust, and/or fire or explosion	System degradation or render inoperative	Loss of redundancy, possible spacecraft damage, possible crew loss	Operate solenoid valves.

Table 26. Command Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Rocket engines (Cont)				External leak	56, 57, 58 and postlanding	Eventual loss of reactant, possible corrosive action and fire	Eventually render inoperative, possible corrosive action and fire	Loss of redundancy, possible corrosive action and fire, possible crew loss	Operate solenoid valves.

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SERVICE MODULE REACTION CONTROL SUBSYSTEM

Progress accomplished during the past quarter on the service module reaction control subsystem (RCS) consisted of (1) evaluation of the presently proposed quad subsystem (2) a modified quad subsystem, and (3) completion of a subsystem failure effects analysis. As a result of these analyses, it has been determined that the present quad and the proposed modified quad meets the required reliabilities. Table 27 presents a summary of the results based on component required reliability rates and component state-of-the-art failure rates.

Table 27. Reaction Control Subsystem Component Reliability Comparison

Configuration	Mission Success (3 out of 4)	Crew Safety (2 out of 4)
Based on Component Required Reliability Rates		
Quad	0.999732	0.999989
Modified quad	0.999675	0.999971
Required	0.99941	0.999970
Based on Component State-of-the-Art Failure Rates		
Quad	0.996337	0.999398
Modified quad	0.994464	0.999013
Required	0.99941	0.999970

CONFIGURATION ANALYSIS

Subsection Description

During the previous quarter, a reliability analysis was performed on the service module reaction control system. The purpose of the analysis was to investigate modifications proposed for the present quad system so that the exact magnitude of possible increased reliability could be determined. The modifications considered were (1) the physical isolation of the reaction control engines in order to minimize the extent of the damage

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in the event of an explosive engine failure, and (2) the addition of two solenoid valves in each quadrant to provide for a more selective engine shut-down capability. If an engine valve should fail in the present quad system, it would be necessary to shut down four engines. The proposed modifications would necessitate the shut down of only two engines if an engine valve should fail. Figure 25, following the mathematical models, shows the difference in the engine configuration for the present and the modified system.

Comparison of results for the quad system and modified systems are presented in Table 27. The results of the analysis indicated that the quad system was more reliable than the modified system for all cases and criteria considered. One reason why the modified system failed to have a higher reliability than the quad system was because additional hardware was required to provide more selective engine shut-down capability. The opinion is that the modifications would be detrimental to the system from a reliability and weight viewpoint. The evaluation of subsystem crew safety is based on results of a derived density function similar to that shown in the abort criteria section. Table 29, following the mathematical models, presents a list of the component failure rates used in this study. The mathematical models used in the analysis are presented in the following table.

Table 28. Quad System Mathematical Models

REGULAR QUAD

(Two of Four for Minimum Control)

MISSION SUCCESS

$$R_{\text{mission success}} = \left[R^4_{\text{module}}\right]\left[R^4_{2\ \text{module}} + 4R^3_{2\ \text{module}}\right.$$
$$(1 - R_{2\ \text{module}})\, R^2_{\text{solenoid}} + 6R^2_{2\ \text{module}}$$
$$\left.(1 - R_{2\ \text{module}})^2\, R^4_{\text{solenoid}}\right] + \left[4R^3_{\text{module}}\right.$$
$$\left.(1 - R_{1\ \text{module}})\, R^2_{\text{solenoid}}\right]\left[R^3_{2\ \text{module}}\right.$$
$$\left. + 3R^2_{2\ \text{module}}\,(1 - R_{2\ \text{module}})\, R^2_{\text{solenoid}}\right]$$

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Table 28. Quad System Mathematical Models (Cont)

MODIFIED QUAD

(Two of Four for Minimum Control)

MISSION SUCCESS

$$
\begin{aligned}
R_{\text{mission success}} = {} & \left[R_{1\ \text{propellant supply}}^{4}\ R_{1\ \text{engine subsystem}}^{8}\right] \\
& \Bigg\{\left[R_{2\ \text{propellant supply}}^{4}\ R_{2\ \text{engine subsystem}}^{8}\right] \\
& + \left[4\ R_{2\ \text{propellant supply}}^{3}\ R_{2\ \text{engine subsystem}}^{8}\ \left(1 - R_{2\ \text{propellant supply}}\right) R_{\text{solenoid valve}}^{4}\right] \\
& + 6\left[R_{2\ \text{propellant supply}}^{2}\ R_{2\ \text{engine subsystem}}^{8}\ \left(1 - R_{2\ \text{propellant supply}}\right)^{2} R_{\text{solenoid valve}}^{8}\right] \\
& + 8\left[R_{2\ \text{propellant supply}}^{4}\ R_{2\ \text{engine subsystem}}^{7}\ \left(1 - R_{2\ \text{engine subsystem}}\right) R_{\text{solenoid valve}}^{2}\right] \\
& + 28\left[R_{2\ \text{propellant supply}}^{4}\ R_{2\ \text{engine subsystem}}^{6}\ \left(1 - R_{2\ \text{engine subsystem}}\right)^{2} R_{\text{solenoid valve}}^{4}\right] \\
& + 32\left[R_{2\ \text{propellant supply}}^{3}\ R_{2\ \text{engine subsystem}}^{7}\ \left(1 - R_{2\ \text{propellant supply}}\right)\left(1 - R_{2\ \text{engine subsystem}}\right) R_{\text{solenoid valve}}^{6}\right] \\
& + \text{Three Failure (Negligible)}\Bigg\} \\
& + 4\left[R_{1\ \text{propellant supply}}^{3}\ R_{1\ \text{engine subsystem}}^{8}\ \left(1 - R_{1\ \text{propellant supply}}\right) R_{\text{solenoid valve}}^{4}\right]
\end{aligned}
$$

Table 28. Quad System Mathematical Models (Cont)

$$
\begin{aligned}
&\Bigg\{\Big[R_2^3 \text{ propellant supply } R_2^8 \text{ engine subsystem}\Big] \\
&+ 8\Big[R_2^3 \text{ propellant supply } R_2^7 \text{ engine subsystem} \\
&\quad (1 - R_{2 \text{ engine subsystem}})\, R^2_{\text{solenoid valve}}\Big] \\
&+ 3\Big[R_2^2 \text{ propellant supply } R_2^8 \text{ engine subsystem} \\
&\quad (1 - R_{2 \text{ engine subsystem}})\, R^4_{\text{solenoid valve}}\Big] \\
&+ \text{Three Failure (Negligible)}\Bigg\} \\
&+ \Big[8\, R_1^4 \text{ propellant supply } R_1^7 \text{ engine subsystem} \\
&\quad (1 - R_{1 \text{ engine subsystem}})\, R^3_{\text{solenoid valve}}\Big] \\
&\Bigg\{\Big[R_2^4 \text{ propellant supply } R_2^7 \text{ engine subsystem}\Big] \\
&+ 4\Big[R_2^3 \text{ propellant supply } R_2^7 \text{ engine subsystem} \\
&\quad (1 - R_{2 \text{ propellant supply}})\, R^4_{\text{solenoid valve}}\Big] \\
&+ 7\Big[R_2^4 \text{ propellant supply } R_2^6 \text{ engine subsystem} \\
&\quad (1 - R_{2 \text{ engine subsystem}})\, R^2_{\text{solenoid valve}} \\
&+ \text{Three Failure (Neglibible)}\Big]\Bigg\}
\end{aligned}
$$

Failure Effects Analysis

Table 30 at the end of this section presents results of the failure effects analysis completed on the service module RCS components. A preliminary failure effects analysis was previously conducted, which formed the basis for the logic block diagram shown in Figure 15 of SID 62-557-3, "Third Quarterly Reliability Status Report," 31 October 1962. The results of the failure effects analysis presented herein show that previously

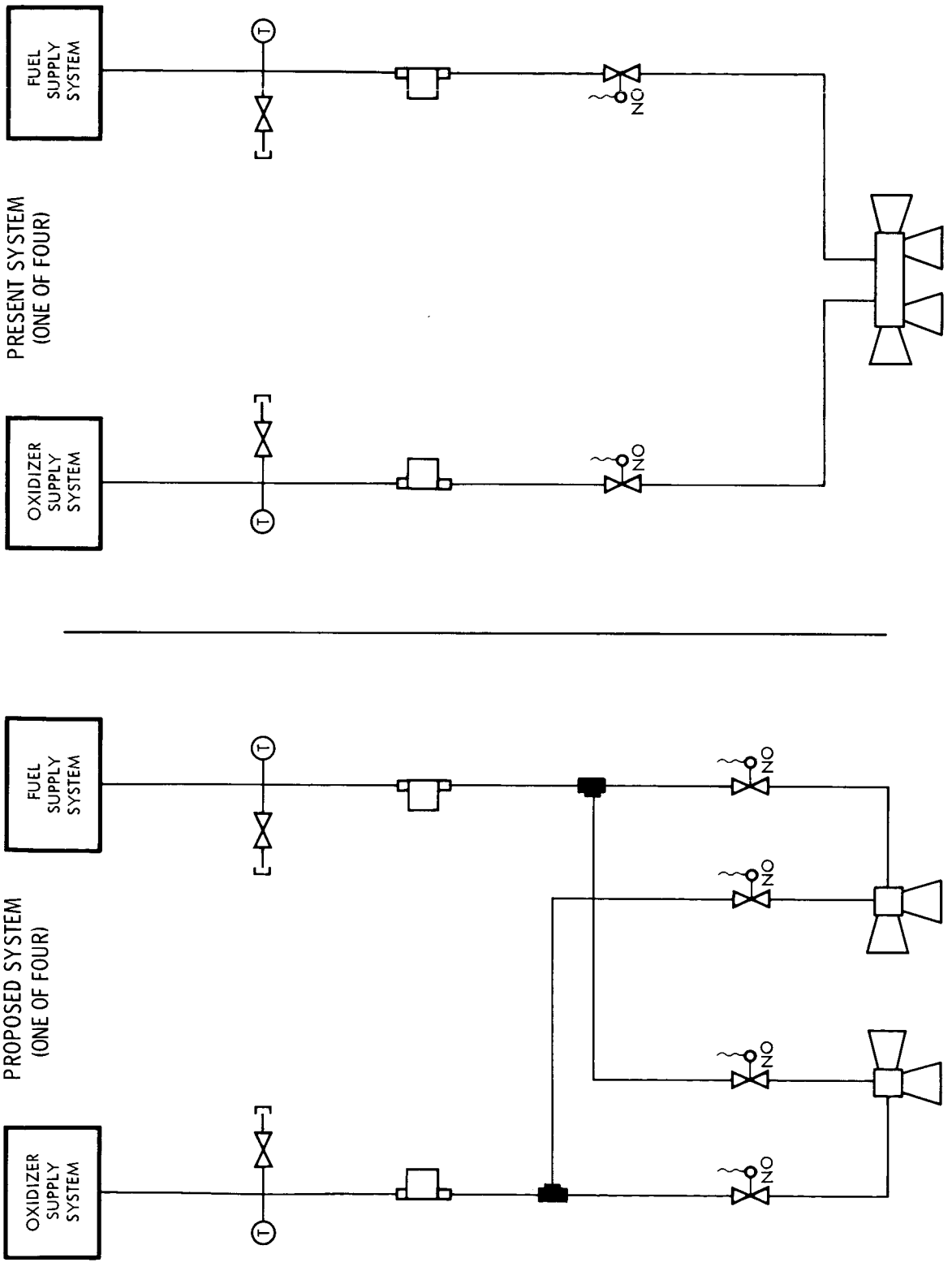

Figure 25. Service Module Reaction Control System Quad Schematic

Table 29. Service Module Reaction Control System Component Failure Rates

Components	Operating Time (hr or cycle)	Apportioned Failure Rates ($\times 10^{-6}$)	Apportioned Reliability	Estimated Failure Rates ($\times 10^{-6}$)	Estimated Reliability
Helium tank	336	0.178	0.999940	0.080	0.999973
Fill and drain valve	336	0.139	0.999953	0.139	0.999953
Solenoid valve	1 cycle	11.0	0.999989	11.0	0.999989
Regulator	336	2.976	0.999000	7.500	0.997480
Check valve	336	2.976	0.999000	2.300	0.999227
Relief valve	336	3.919	0.998683	3.919	0.998683
Burst disk	1 cycle	1.0	0.999999	1.0	0.999999
Filter	336	0.300	0.999899	0.300	0.999899
Vent valve	336	0.139	0.999953	0.139	0.999953
Oxidizer tank	336	0.297	0.999900	1.601	0.999462
Fuel tank	336	0.297	0.999900	1.601	0.999462
Flow meter	336	1.0	0.999664	1.0	0.999664
Cap	336	0.011	0.999996	0.011	0.999996
Oxidizer valve	336	1.636	0.999450	4.940	0.998340
Fuel valve	336	0.744	0.999750	4.940	0.998340
Injector head	0.1388	5760.0	0.999200	43200.0	0.994000
Thrust chamber	0.1388	10,080.0	0.998600	28800.0	0.996000
Engine	336	8.928	0.997000	39.464	0.986740

defined system logic is still valid. The equations shown herein present the mathematical logic of the system based on the results of this failure effects analysis.

TEST RESULTS

The rocket engine development test program and materials evaluation testing is in progress at Marquardt. Included in this testing were engines, thrust chambers, solenoid valves, and thrust chamber material coatings. A 1-million-cycle life test was conducted on a single-engine solenoid valve during this reporting period. Leakage tests were performed at specified intervals throughout this test.

Marquardt is continuing to use statistically designed test plans in its development test program and is using statistical methods to evaluate thrust chamber coating thickness. This information is being used as an aid in designing and assessing reliability.

Items of major importance in the test program for the next quarter include: (1) continued development testing of the engine and thrust chamber and minimum impulse testing using dual-coil solenoid valves, and (2) compilation of a list of all test instrumentation with rated accuracy.

Effects of Qualification Program Reorientation

As a result of the recent qualification program redirection, a total of 12 Marquardt service module RCS reaction control system engines and a total of 68 Rocketdyne command module RCS engines were deleted from the test program. The maximum number of components within the propellant-feed system was reduced from 12 to 7. The actual number of components allowed for test is based upon criticality (the effect that its failure has on crew safety). The requirements for sand and dust and salt spray tests were also deleted.

SUBCONTRACTOR MANAGEMENT

Marquardt Corporation, manufacturer of the Apollo service module RCS engines, has accomplished the reliability tasks as described herein during the past quarter.

Part II of the reliability program plan describing the statistically designed development test plans and reliability demonstration test plans was completed. Because of the redirected reliability testing concept, the program plan will require revision.

A formal design review of the propellant valves has been completed and thoroughly documented. Further design reviews of the injector and thrust chamber are scheduled for the forthcoming quarter.

Concurrent with the design review, a detailed failure mode analysis of the propellant valves was performed. Similar analyses will be made of the injector and thrust chamber during the next quarter.

During the ensuing quarter, completion of a failure reporting and failure analysis system are contemplated. A training program is also being developed and is scheduled for completion during the next quarter.

PLANNED ACTIVITIES

Additional subcontractors will be involved in the service module RCS so that coordination with more suppliers than Marquardt will be necessary.

Continued analyses will be made to determine the degree of component reliability improvement required. The results of the reliability analyses on the various modifications will be presented in the next quarterly progress report.

Items of major importance in the test program for the next quarter include (1) continued development testing of the engine and thrust chamber and minimum impulse testing using dual coil solenoid valves, and (2) compilation of a list of all test instrumentation with its rated accuracy.

Table 30. Service Module Reaction Control Subsystem Failure Effects Analysis

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Helium tank (1)			Store presurization gas	Leak at connection, excessive vibration	1 through 55	Loss of reaction control system module use	Mission degradation (minor)	None	None
Solenoid valve (crew operate) normally open (2)			Emergency shutoff of pressure to propellant supply in case of regulator failure	Fail to close, shorted or broken solenoid coil wire, binding of armature leak , damaged seat	1 through 55	None overpressurization causing relief valve to operate, loss of helium if both valves fail	Mission degradation (minor)	None	None
Helium fill and drain valve with cap (3)			Fill or drain helium	Leak, excessive vibration (flight)	1 through 55	Loss of helium and reaction control system module use	Mission degradation (minor)	None	None
Regulator (4)			Regulate propellant feed pressure	Fail to open, binding, broken spring, fail to close, binding, contamination, Lock up (leak), damaged seat, contamination	1 through 55	None if one path fails loss of reaction control system module use if both paths fail None if one/one path fails, overpressurization if both paths fail None if one/one path fails, overpressurization if both paths fail	Mission degradation (minor)	None	Actuation of one/both helium solenoid shut-off valves
Check valve (4)			Prevent mixing of fuel and oxidizer vapors in case of positive explusion means failure	Fail to open, binding, fail to close, broken spring, contamination, damaged seat	1 through 55	None if one fails; no pressurization of fuel or oxidizer tank if two in parallel fail None if one or two in parallel fail; possible explosion if two or more in series fail	Mission degradation (minor) Possible da damage to service module (major/ critical)	None Abort	None

Table 30. Service Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Relief valve assembly (relief valve, filter and burst disconnect) (2)			Provide relief of pressure in event of overpressure conditions	Fail to open binding Fail to close, broken spring, contamination, binding	1 through 55	Possible rupture of lines, loss of propellant pressure, thus loss of reaction control system module Possible rupture of lines, loss of propellant pressure, thus loss of reaction control system module	Mission degradation (minor)	None	None
Positive expulsion fuel/oxidizer tank (1 ea.)			Contain fuel/oxidizer in usuable condition	Expulsion device leaks, rupture Tank leak at connection, excessive vibration	1 through 55	Possible loss of engine start capability Loss of propellant thus loss of reaction control system module	Mission degradation (minor)	None	None Close both helium shut-off valves
Fuel/oxidizer fill and drain valve with cap (2)			Fill or drain fuel/oxidizer	Leak, excessive vibration (flight)	1 through 55	Loss of fuel/oxidizer thus loss of reaction control system module	Mission degradation (minor)	None	None
Fuel/oxidizer solenoid valve (crew operated) normally open (1 ea.)			Emergency shut-off of propellant to engines, in event of propellant valve open failure	Fail to close, shorted or broken solenoid wire, binding armature Leak, damaged seat Spontaneous actuation, radio frequency noise	1 through 55	Loss of propellant, thus loss of reaction control system module	Mission degradation (minor)	None	None

Table 30. Service Module Reaction Control Subsystem Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Fuel/oxidizer valve (4 ea.)			Provide fuel and oxidizer to reaction control system engine in required amounts	Fail to open, short or broken solenoid wire, armature binding	1 through 55	Loss of use of reaction control system module	Mission degradation (minor)	None	Utilize manual operation circuit
						Loss of use of reaction control system module	Mission degradation (minor)	None	Utilize manual operation circuit
				Fail to hold open, hold coil failure, signal failure, fail to close, broken spring, binding Leak, damaged seat, contamination, spontaneous actuation, radio frequency noise		Loss of propellant, loss of reaction control system module	Positive expulsion loss of quadrant mission degradation	Single quadrant loss is not sufficient for abort, two quadrant loss then abort	Detect failure and actuate fuel or oxidizer solenoid shut-off valve
						Loss of propellant, loss of reaction control system module	Positive expulsion loss of quadrant mission degradation	Single quadrant loss is not sufficient for abort, two quadrant loss then abort	Detect failure and actuate fuel or oxidizer solenoid shut-off valve
						Excessive use of propellants	Mission degradation (minor)	None	None
Injector (4)			Provide proper propellant flow pattern to combustion chamber	Leakage of propellants	6 through 55	Improper engine operation	Mission degradation (minor)	None	None
Thrust chamber (4)			Provide confined combustion area and provide desired thrust characteristics	Fracture, thermal stress excessive erosion, damaged coating	6 through 55	Loss of engine use		None	None
						Loss of engine use		None	None
						Loss of engine use		None	None
				Burn-out, recrystalized material Explosion, throat restriction Hot spot, contamination		Possible loss of reaction control system module		Possible abort	Abort
						Loss of engine		None	None

SERVICE MODULE SERVICE PROPULSION SUBSYSTEM

The reliability effort for this quarter on the service propulsion system (SPS) covers a revision of the reliability allocations on the Aerojet-General engine subsystem, a detailed failure effects analysis, a comparative analysis of various propellant utilization systems, and an analysis of a parallel feed system for the SPS propellants.

The service propulsion system current state of the art and required reliability was calculated and the results are shown in Table 31. The service propulsion system logic diagrams shown in the Third Quarterly Progress Report, NA 62-557-3, are still valid.

Table 31. Service Propulsion System Reliability

System	State of the Art	Required
S/M - SPS	0.9646	0.999770

A major effort during this quarter has been the revision and updating of procurement specifications and the evaluation of proposals for service propulsion system components and subsystems.

CONFIGURATION ANALYSIS

Subsystem Description

The service module service propulsion system is comprised of the following subsystems: helium pressurization, propellant feed, propellant utilization, and engine. The purpose of the service propulsion system is to provide thrust vector orientation and ΔV for midcourse corrections, lunar orbit injection, and transearth injection.

The helium pressurization subsystem supplies regulated helium to the propellant feed system and isolates the fuel and oxidizer vapors during coast.

The propellant feed system, in turn, supplies pressurized fuel to the engine control valves and propellants to the rocket engine.

The propellant utilization subsystem allows the crew to adjust the fuel to oxidizer mixture ratio, so that the propellants will be maintained at the proper ratio and reduce the residual propellants to a minimum.

Results of Analysis

The mathematical model for the service module propulsion logic network is as follows:

$$R_{system} = R_1^2 \cdot \left[1 - (1 - R_2)(1 - R_3)\right] \cdot \left[1 - (1 - R_4, R_5^2)^2\right] \left[1 - (1 - R_6^2)^2\right] \cdot \left[1 - (1 - R_8, R_9)(1 - R_7)\right] \cdot \left[1 - (1 - R_{10})(1 - R_{11})\right] \cdot R_{12}^2 \cdot \left[1 - (1 - R_{13})(1 - R_{14})\right] \cdot R_{15} \cdot R_{16} \cdot R_{17} \cdot \left[1 - (1 - R_{18}^2)^2\right] \cdot \left[1 - (1 - R_{19} \; R_{21})(1 - R_{20})\right] \cdot \left[1 - (1 - R_{22})(1 - R_{23})\right] \cdot R_{24}^2 \cdot R_{25} \cdot \left[1 - (1 - R_{26})(1 - R_{27})\right] \cdot R_{28} \cdot R_{29} \cdot R_{30} \cdot \left[1 - (1 - R_{31}^2 \cdot R_{32}^2 \cdot R_{33}^2 \cdot R_{34}^2)^2\right] \cdot \left[1 - (1 - R_{35}^2)\right] \cdot R_{36} \cdot R_{37} \cdot \left[1 - (1 - R_{38})^2\right]^2 \cdot R_{39}$$

(The reliabilities represented by R_1, R_2 ... R_{39} are the system components shown in Figure 11 of the Third Quarterly Reliability Status Report, SID 62-557-3

Service propulsion system component failure rates used to calculate the system current state-of-the-art and required reliability are shown in Table 32.

The effects of removing the automatic propellant utilization system from the service module service propulsion system was re-evaluated. A comparison of reliabilities using the two systems is shown in Table 33.

The reliability requirements of 26×10^{-6} must be met by whichever system is selected. A propellant quantity indicating system is a minimum requirement to assist the crew in abort decisions.

Table 32. Service Module - Service Propulsion System Component Failure Rate Data

Component	Mean State-of-the-Art Probability of Failure x 10^{-6}	Apportioned Probability of Failure x 10^{-6}
Helium tank	26.88	5
Fill and drain and cap	1	1
Solenoid valve	11	11
Regulator	2,520	1000
Check valve	772.8	1000
Relief valve	1,317.12	1000
Burst disk	100	100
Filter	100.8	5
Oxidizer tank	53.8	14
Fuel tank	53.8	14
Propellent utilization system		26
Vent valve	1,318.4	1000
Heat exchanger	60	18
Rocket engine	30,288	100

Table 33. Automatic Versus Manual Propellant Utilization Reliability Comparison

System	System Reliability	Difference in Reliability	Estimated Residual Propellants	Difference in Residuals	Ratio 2:4	Estimated System Weight	Estimated Residual Weight of System
Automatic propellant utilization	$26X10^{-6}$		100			75	175
Manual propellant utilization	$20X10^{-6}$	$6X10^{-6}$	200	100	$6X10^{-8}$	65	265
Quantity indicating	$5X10^{-6}$	$21X10^{-6}$	450	350	$6X10^{-8}$	50	500

Deletion of the flow control requirements altogether would require that excess propellants be loaded to account for the variables introduced by engine engine, oxidizer fuel ratio, engine and propellant feed system mismatch and possible change in P during the mission. The automatic propellant utilization system is designed to maintain the residual propellants at 100 pounds. The propellant quantity gaging system alone would need an excess of 450 pounds of excess propellants. The reduced complexity of the system would reduce the probability of failure to 5 per 10^6 missions. On this basis, the ratio of system reliability to increase in residual propellants weight is 6×10^{-8} per pound of residual propellants (Table 33). The control and valve comprise approximately 1/3 of the total propellant utilization system weight. The manual flow control system is more complicated than the gaging system, with an apportionment of 20 per 10^6 missions (Table 33). A manual system, designed to maintain the residuals at 100 pounds over the automatic system, has a ratio of system reliability to increase in residual of 6×10^{-8} per pound of residual propellant, with a decrease in propellant utilization system weight of 15 pounds. The difference in residual weight of the Apollo spacecraft effected by use of the manual propellant utilization system over quantity indicating gaging only would be approximately 235 pounds.

The service module SPS parallel feed system approach to CG control after zero-g environment will be satisfactory, even though slight leakage past the tank outlet check valves may occur. Major loss of propellant from one tank to the other is critical; therefore, the parallel feed system must have parallel check valves at the outlet of each tank for reliability. A second approach using a floating-ball, antitransfer device was reviewed for test requirements to demonstrate a probability of failure of 2×10^{-6}.

The logic diagrams shown as Figures 11 and 13 in the previous Quarterly Reliability Progress Report, NA 62-557-3, were based on a preliminary failure effects analysis. The basic logic has not changed in this quarter.

Summary of Failure Effects Analysis

The individual components of the service propulsion system were analyzed and the failure modes determined. The effects of redundancy in eliminating most critical failures is shown in the failure effects analysis, Table 34 at the end of this section. Other components which could cause catastrophic failure and where redundancy is not practical show up as affecting the Apollo spacecraft. These components, however, require a high reliability, and the current state-of-the-art and required reliability are nearly the same.

TEST PROGRAM

The development program and materials evaluation testing is continuing. Significant results observed during these tests follow:

1. A series of firings accumulated a total of 900 seconds on an ablative test chamber without failure. This duration is equal to design time and is 150 seconds in excess of the required specification time.

2. A five-baffle injector was test fired five times, demonstrating the injector restart capability.

3. Subscale engine testing at AEDC, Tullahoma, Tennessee, is complete. The results of those tests, attitude performance and material evaluation, are in process.

Items of major importance scheduled for the next quarter are:

1. Continued development, including more subscale engine testing at AEDC.

2. Initiation of lightweight injector and chamber testing at Sacramento

Propellant seal and o-ring compatability tests and methods of brazing and welding tube connections are being evaluated at S&ID's Engineering Development Laboratories. The fixture to be used for propellant feed system concept evaluation is virtually complete, and testing is scheduled to start next quarter at S&ID.

Development testing of propellant and pressurization components will commence during the next quarter.

Service Propulsion Rocket Engine

Aerojet revised the qualification-reliability test plan for the service propulsion rocket engine. However, as a result of recent program redirection, this plan will be incorporated in the Aerojet overall test plan. The service propulsion system procurement specification (MC 901-0009) was also revised as a result of the program redirection.

The development program and materials evaluation testing is continuing. The following items are significant results observed during these tests.

1. A total of 900 seconds firing was accumulated on an ablative test chamber.

2. A five-baffle injector was test fired five times, demonstrating the injector restart capability.

3. Subscale engine testing is in progress at Tullahoma, Tennessee, to evaluate altitude performance capability.

Effects of Qualification Program Reorientation

As a result of the recent qualification program redirection, a total of 15 of the Aerojet rocket engines and greater than half of the rocket engine components that had been scheduled for test were deleted from the test program. The maximum number of components within the propellant feed system was reduced from 12 to 7. The actual number of components allowed for test is based upon criticality of the item (the effect that its failure has on crew safety). The requirements for sand and dust and salt spray tests were deleted.

PROBLEM AREAS

The required improvement over the estimated state of the art for the rocket engine subsystem is approximately 154 to 1. This degree of improvement in a critical assembly will require very careful monitoring of the overall program. The current state of the art was based on engine systems which are not man-rated.

PLANNED ACTIVITIES

An extensive effort to revise all procurement specifications is in progress. Specifications which have been released are being revised to incorporate criticality requirements and the latest reliability apportionments. Evaluation of proposals and selection of vendors for service propulsion system equipment will be continued, with reliability requirements coordinated with vendors. Items of major importance scheduled for the next quarter are:

1. Continued service propulsion rocket engine development, including more subscale engine testing, injector and ablative chamber testing

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2. Initiation of injector and chamber testing at Sacramento, California

3. Breadboard testing scheduled at Aerojet to evaluate the entire thrust chamber assembly

Table 34. Service Module Service Propulsion System Failure Effects Analysis

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Helium tank			Store pressurizing gas	Leakage tank wall fittings	Boost trans-lunar, lunar orbit, trans-earth	Loss of Pressurizing gas	Loss of available thrust	Mission degradation	Abort
				Over pressure		Rupture tank	Loss of subsystem	Mission abort	Abort
Helium fill and drain valve			Fill and drain helium	Leakage	Boost trans-lunar, lunar orbit, trans-earth	Loss of gas	None redundant with cap	None	None
Cap			Seal helium fill and drain valve	Leakage	Boost trans-lunar, lunar orbit, trans-earth	Loss of gas	None, redundant fill and drain valve	None	None
Solenoid valve normally open			Isolate failed open series regulator valves	Open	Boost trans-lunar, lunar orbit, trans-earth	Allow over-pressurization of propellant tanks	Vent the pressure through relief valve	None, required to operate only after double failure	None
				Closed		None	None	None-redundant path for regulation	None
Regulator			Regulate propellant tank pressure	Open	Boost trans-lunar, lunar orbit, trans-earth	None, redundant with another regulator	None	None	None
				Closed		None	None	None	None
				Burst		Loss of pressurization gas	Loss of available thrust	Mission abort	None

Table 34. Service Module Service Propulsion System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Check valve			Separate propellant vapors	Leak	Boost trans-lunar, lunar orbit, trans-earth	None, series redundant	None	None	None
				Closed		Redundant system increase in ΔP	Mixture ratio error propellant utilization system correction	None	None
Relief valve			Relieve propellant tank over pressurization	Open	Boost trans-lunar, lunar orbit, trans-earth	Loss of pressurizing gas	Loss of subsystem	Mission abort	Abort
				Closed		Tank over pressurization	Loss of subsystem	Mission abort	Abort
Burst disk			Prevent leakage until system failure causes over pressurization	Leak	Boost trans-lunar, lunar orbit, trans-earth	None, series redundant with relief valve	None	None	None
				Burst pressure high		High tank pressure	Mixture ratio error	None	None
Filter			Catch particals from burst disk and helium system	Blockage	Boost Trans-lunar, lunar orbit, trans-earth	Propellant tank over pressurization	Mission abort	Mission abort	None
Vent valve			Ground operation vent during fill	Leakage	Boost trans-lunar, lunar orbit, trans-earth	None, series redundant cap	None	None	None
				Open		None, ground operation	None	None	None
				Closed				None	None

Table 34. Service Module Service Propulsion System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Cap			Seal vent valve	Leakage	Boost trans-lunar, lunar orbit, trans-earth	None, series redundant	None	None	None
Propellant tank			Store propellant	Leakage	Boost, trans-lunar, lunar orbit, trans-earth	Loss of propellant or pressurizing gas	Loss of propel-lant	Mission abort	Abort
				Over pressure		None, relief valve	Loss of pres-surizing gas	Mission abort	Abort
Fill propellant drain			Ground operation fill and drain	Leakage	Boost trans-lunar, lunar orbit, trans-earth	None, redundant cap	None	None	None
Cap			Seal propellant fill and drain valve	Leakage	Boost trans-lunar, lunar orbit, trans-earth	None, series redundant with fill and drain valve	None	None	None
Propellant line flex bellows propellant filter			Allow engine gimbaling Filter propellants	Leakage	Boost trans-lunar, lunar orbit, trans-earth	Loss of propellant	None, series redundant bellows	None	None
				Blockage		Excess ΔP	Mixture ratio error	Mission degra-dation	None
Propellant orifice			Balance system ΔP	Blockage	Boost trans-lunar, lunar orbit, trans-earth	Excess ΔP	Mixture ratio error	Mission degra-dation	None
Heat Exchanger			Heat helium	Leakage	Boost trans-lunar, lunar orbit, trans-earth	Loss of pressuri-zation gas	Loss of avail-able thrust	Mission abort	None

Table 34. Service Module Service Propulsion System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Propellant gaging			Continuous-propellant quantity reading	Short	Trans lunar, lunar orbit transearth	None, parallel redundant system	None	None	None
Crew display			Continuous-propellant quantity read-out	Short	Trans lunar, lunar orbit transearth	None, redundant crew display	None	None	None
Manual flow control			Intermittent flow control	Motor	Trans lunar, lunar orbit transearth	None, redundant	None	None	None
				Position indicator		None, redundant			Manual correction
Solenoid valve			Engine start valve	Leakage	Boost (abort only)	None, series redundant	None	None	None
				Fail to open	Trans lunar, lunar orbit, transearth	None, parallel redundant	None	None	None
				Fail to close		None, series redundant	None	None	None
Valve actuator			Open engine valves	Leakage	Boost (abort only) translunar, lunar orbit, transearth	None, bellows redundant	None	None	None
				Open		None parallel redundant	None	None	None
				Close		None series redundant	None	None	None

Table 34. Service Module Service Propulsion System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Fuel valve				Leakage	Boost (abort only) trans-lunar, lunar orbit, trans-earth	Series none, redundant	None	None	None
				Open		None, parallel	None	None	None
				Close		None, series	None	None	None
Oxidizer valve				Leakage	Boost (abort only) trans-lunar, lunar orbit, trans-earth	Series none, redundant	None	None	None
				Open		None, parallel	None	None	None
				Close		None, series	None	None	None
Check valve actuator vent				Leakage	Boost (abort only) trans-lunar, lunar orbit, trans-earth	Series none, redundant	None	None	None
				Fail to open		None, parallel	None	None	None
Injector				Orfice plugged	Boost (abort only) trans-lunar, lunar orbit, trans-earth	None	None	None	None
				Burnout		Loss of thrust chamber	Loss of thrust	Mission abort	None
Gimbal bearing				Flexture	Boost (abort only) trans-lunar, lunar orbit, trans-earth	None, redundant	None	None	None

Table 34. Service Module Service Propulsion System Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Gimbal actuator ball screw				Stuck	Boost (abort only) trans lunar, lunar orbit, trans-earth	Loss of thrust vector control	Loss of thrust vector control	Mission abort	None
Motor				Open	Boost (abort only) trans lunar, lunar orbit, trans-earth	None, redundant loop	None	None	None
				Shorted					
				Binding					
Clutches				Open		None, redundant loop	None	None	None
				Shorted					
				Leakage					
Gear train				Mesh		None, redundant loop	None	None	None
				Jammed					
				Fatigue					
Chamber and nozzle exit				Excessive erosion	Boost (abort only) trans-lunar, lunar orbit, trans-earth	Low pc	Reduction in total impulse	Mission abort	Abort

COMMAND MODULE HEAT SHIELD

The major reliability effort for the heat shield has been in developing analytical techniques, in the preparation of qualification test procedures and in defining and updating environmental criteria.

Until a detailed reliability analysis can be completed, it is assumed that the heat shield will meet its reliability goal of 0.99995. In order to obtain a valid reliability figure, the concept of basing reliability assurance on measurements of material behavior in tests of sample materials exposed to simulated operational environments has been followed. These tests would include thermal, vibrational, shock, and other loadings as stress factors, acting alone and in combination.

System reliability is, of course, a function of operational performance. Operational performance can be related to destructive tests conducted in the laboratory within the state of the art of simulation capability. Therefore, by evaluating the performance of the samples with respect to the backface temperature and other criteria under the simulated environments included in the qualification-test program and other development tests, a measure of the heat-shield reliability will be obtained.

One facet of the overall reliability program is the derivation of a method of quantitatively assessing the inherent reliability of the Apollo heat shield and subsequent calculations of this value. The statistical approach in determining the reliability is presented herein.

CONFIGURATION ANALYSIS

The methods, depth, and overall adequacy of analysis techniques used by the subcontractor in Apollo heat shield developmental efforts will be subject to review and approval by NAA.

The statistical approach in determining the reliability will be one which will consider the distribution of backface temperature measurements and involve the computation, through the use of well-known statistical techniques, of the probability that the temperature will not exceed the specified acceptable backface temperature limit (while maintaining structural integrity). Specifically, what will be done is as follows:

1. All data associated with a measure of backface temperature during development tests, first, will be evaluated for inclusion

in the population of data observations and, second, be plotted in the distribution of these observations.

2. The qualification test results will also be evaluated at the conclusion of the testing as described in (1).

3. The best-estimate will be calculated, using the following procedure.

We wish to calculate probability ($T < 600$ F), with confidence level (γ).

Let $t_o = \dfrac{(600 - \overline{T})\ \sqrt{N}}{\sigma}$.

Let $\theta = \dfrac{|\ t_o\ |}{\sqrt{2f}}$, where $f = \hat{N} - 1$.

If $\theta > 0.75$, calculate

$$k_1 = \frac{1}{\sqrt{1 + t_o^{\,2}/2f}}$$

and if $\theta < 0.75$, calculate

$$k_2 = \frac{t_o}{\sqrt{2f}} \left[1 + \frac{t_o^{\,2}}{2f} \right]^{-1/2}$$

Entering the Johnson and Welch tables for the noncentral "t" distribution, [1] look up

$$\lambda = \lambda \left(f,\ t_o,\ 1 - \gamma,\ k_1 \text{ or } k_2 \right);$$

let

$$Z_R = \frac{t_o - \lambda \sqrt{1 + t_o^{\,2} / 2f}}{\sqrt{\hat{N}}}$$

---

[1] Johnson, N. L., and B. L. Welch, Applications of the Non-Central t-distribution, Biometrika, 31 (1939). p. 362-389.

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Entering a table of the normal probability distribution find R such that

$$R = \int_{-\infty}^{Z_R} \phi(t)\, dt,$$

Where $\phi(t)$ is the normal distribution.

4. An appropriate confidence level commensurate with the sample size of data observations upon which the reliability statement is based will be determined.

Assuming conservative backface temperature results coupled with required structural properties, the number of data observations obtained from this phase of the program will be sufficient to permit a statement of the reliability consistent with the apportioned value.

The foregoing technique will not only provide the basis for computing the reliability, but also permit the comparative analysis of various thicknesses, materials, bonds, environments, etc., as they affect backface temperature. This information will then be utilized in the design process and lead to effective tradeoff decisions.

TEST PROGRAM

Matrix test designs have been formulated for the sequential tests (ablative) and the nuclear radiation tests. The planned design of experiments is undergoing review.

Analyses of heat of ablation versus air enthalpy experimental data have been completed for four materials. These analyses consisted of estimating and comparing the essentially linear relationships of the representative groups of experimental materials. Defined prediction limits were derived for those groups exhibiting essentially equivalent heat of ablation versus air enthalpy properties.

In association with an experimental testing procedure, a statistical procedure has been developed to derive simultaneous confidence levels of specific heat and thermal conducting properties of Apollo ablative materials.

Approximately 100 physical property tests have been performed on the ablator and bond material. These preliminary tests have defined ablator tensile and compressive properties and bond shear and tensile characteristics.

Important factors influencing the batch-to-batch consistency of Avcoat 5026-39 have been determined. These are (1) surface activity of the phenolic microbaloons (as influenced by the moisture content), (2) surface finish on fibers, (3) melting point of the Harcure "A" (as influenced by acid content).

A program has been initiated to develop simple and reliable techniques for direct application of the Apollo heat-shield material to the structure.

Avcoat 5026-39 specimens were tested under vacuum at room temperature and vacuum at +350 F (utilizing infrared heating of one side of specimen). In each test, the initial level of vacuum was 10^{-6} torr. Outgassing characteristics were measured as follows:

Temp (°F)	Time (hr)	Average Pressure (torr)	Weight Loss (gm/cm^2)
Room	3	10^{-6}	0.0014
350	4	10^{-4}	0.0021

Entry-heating and cold-soak tests of fiberglass-filled honeycomb ablator attached to a steel sandwich structure have been accomplished. The backface temperature and char formation resulting were very mild.

Cold-soak and resonant vibration tests of an Avcoat 5026-39 ablator bonded to a stainless steel honeycomb panel were conducted. Visual inspections after test did not reveal any failures.

Cold-soak tests of "cast on" tile specimen cured in place on steel have been completed and no failures occurred.

Heat flux simulation tests were accomplished on an Avcoat 5026-39 ablator, bonded to a PH-15-7 Mo steel sandwich. At the conclusion of the test, it was noted that the char appearance and hardness were good. No failure of the substructure bond was found.

EFFECTS OF QUALIFICATION PROGRAM REDIRECTION

As a result of the redirected effort in the qualification test area, a more realistic qualification test program was defined, and detailed test procedures were prepared. In addition, the following qualification test requirements were deleted: fungus, sand and dust, acoustics, and acceleration. Test program responsibility was delegated to NAA.

PLANNED ACTIVITIES

Continuous effort at S&ID is being directed toward development of analysis techniques in the area of structural reliability. It is expected that any progress in this area will materially aid developmental efforts on the Apollo heat shield.

Efforts are being directed to developing a matrix type test for the solar radiation tests, as well as to investigating the existing nondestructive testing techniques, as applicable to the evaluation of the heat shield material and its assembly to the substructure.

LAUNCH ESCAPE SUBSYSTEM

During the period covered by this report, major effort was directed toward the development phase of the program. Data from development tests are being reduced and will be used where applicable to assess current reliability levels. At present, the estimated system reliability is 0.9916, which is below the required value of 0.999989 as indicated in Table 35.

Table 35. Launch Escape Subsystem
Estimated and Required Reliability

Component	Estimated Reliability*	Required Reliability*
Initiators	0.999	0.999
Launch escape motor	0.993	0.998
Pitch control module	0.9992	0.9998
Tower jettison motor	0.9998	0.99995
Tower structure	0.99999	0.99999
Tower separation mechanics	0.99999	0.99999
System	0.9916	0.999989

*NOTE: These values are based on logic math models and detailed component failure rates presented herein.

It is evident from the foregoing table that many reliability goals have been attained, and it is anticipated that all subsystems will be shown to surpass the required values as additional test data are reduced. The following paragraphs outline accomplishments and goals for each major subsystem of the launch escape system.

CONFIGURATION ANALYSIS

Subsystem Description

The launch escape system is intended for use during the lift-off and boost phases to effect separation of the command module from the spacecraft should an abort condition occur. The system consists of five major subsystems:

1. Launch escape motor, which provides thrust necessary to effect separation of command module from the spacecraft.

2. Pitch control motor, which provides side thrust to establish command module trajectory.

3. Tower jettison motor, which provides thrust to separate tower from command module.

4. Tower structure, which serves as the base for system motors.

5. Tower separation mechanism, which is the mechanism interface between tower and command module.

Logic diagrams and mathematical models (Figures 26 and 27) have been prepared for the normal and abort modes of operation. Subsystem logic, which forms the basis for the logic block diagram formulation and mathematical model derivation, is based on the results of the failure affects analyses in Table 39 at the end of this section.

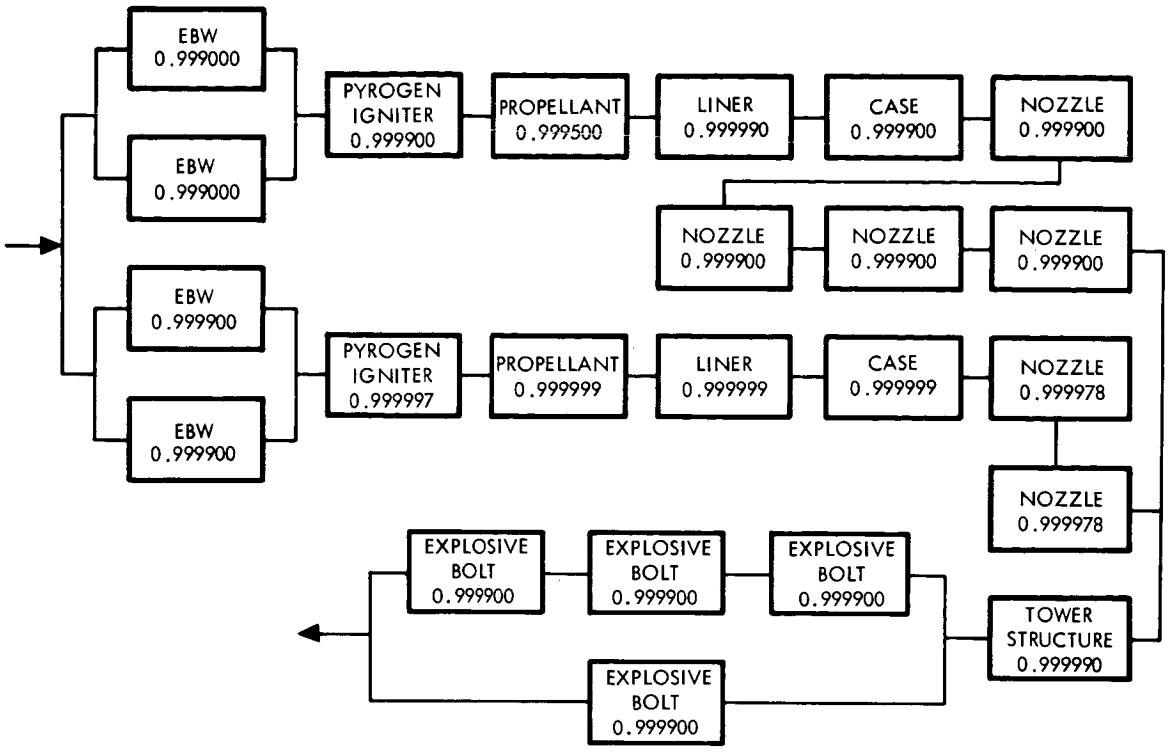
Launch Escape Motor

The launch escape motor provides the thrust to effect separation of the command module from the spacecraft should an abort condition exist.

Table 36 indicates the estimated state of the art and required reliability levels for launch escape motor.

The launch escape motor development phase is up to date. The major achievements during this report period were delivery of motor 1 (inert) to S&ID and the successful test firing of motors 2, 3, and 4.

Detailed reliability evaluations of the engineering drawings and specifications for the launch escape motor were continued. To determine the margin of safety, and thus determine reliability, a study to find the measurements required to perform this task was undertaken. The insulation thickness for the launch escape motor will be measured prior to loading the



NORMAL MISSION MATH MODEL

$$P_{SUCCESS} = \left[P_1 P_2 + P_1 (1 - P_2) + P_2 (1 - P_1)\right] P_3 P_4 P_5$$

WHERE

P_1 = PROBABILITY OF SUCCESS OF TOWER JETTISON MOTOR

$$= \left[P^2_{EBW} + 2 P_{EBW} (1 - P_{EBW})\right] \left[P_{PYROGEN\ IGNITER}\right] \left[P_{PROPELLANT}\right] \left[P_{LINER}\right] \left[P_{CASE}\right] \left[P_{NOZZLE}\right]^2$$

P_2 = PROBABILITY OF SUCCESS OF LAUNCH ESCAPE MOTOR

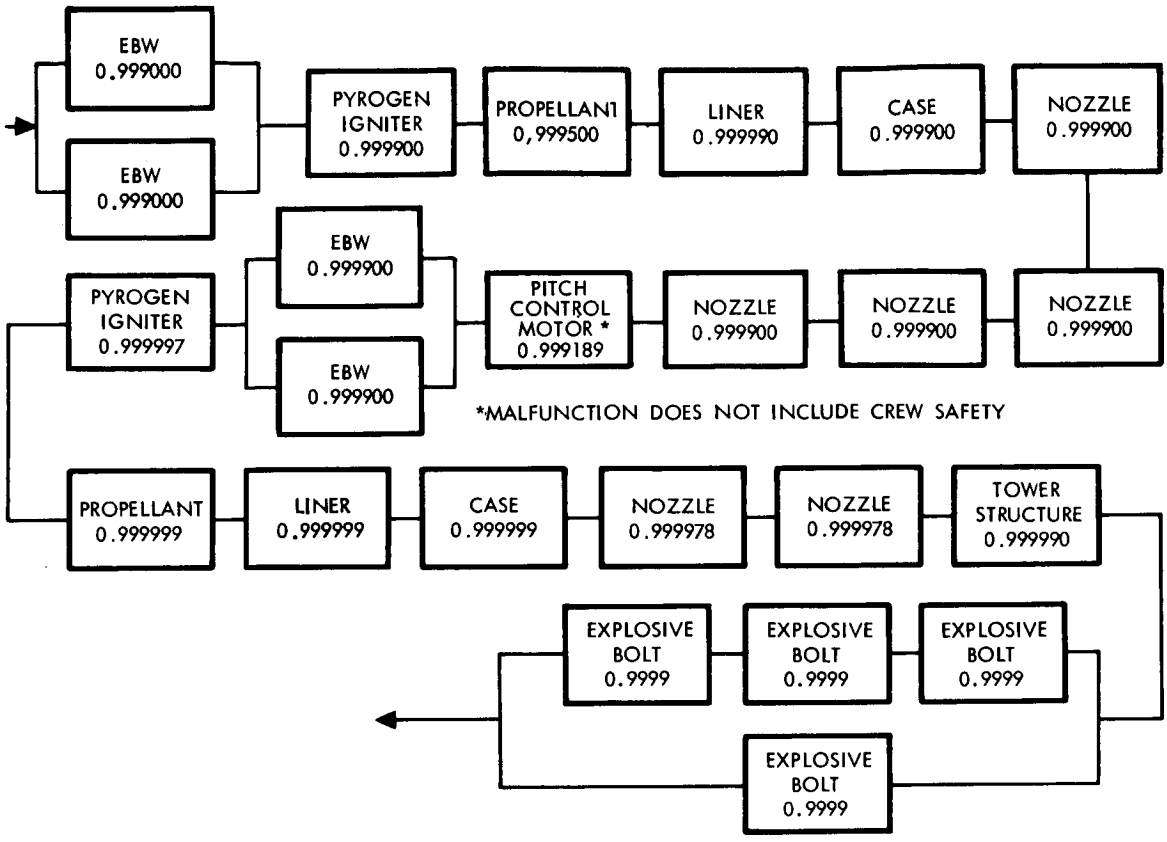
$$= \left[P^2_{EBW} + 2 P_{EBW} (1 - P_{EBW})\right] \left[P_{PYROGEN\ IGNITER}\right] \left[P_{PROPELLANT}\right] \left[P_{LINER}\right] \left[P_{CASE}\right] \left[P_{NOZZLE}\right]^4$$

$$P_3 = P_{TOWER\ STRUCTURE}$$

$$P_4 = P^2_{EBW} P^2_{CUTTER} + 2 \left[P_{EBW} P_{CUTTER} (1 - P_{EBW} P_{CUTTER})\right]$$

$$P_5 = \left[P^4_{BOLT} + 4P^3_{BOLT} (1 - P_{BOLT})\right]$$

Figure 26. Normal Mission Tower Jettison Reliability Logic Diagram



LAUNCH ABORT AND TOWER JETTISON MATH MODEL

$$P_{SUCCESS} = P_1 P_2 P_3 P_4$$

WHERE

P_1 = PROBABILITY OF SUCCESS OF LAUNCH ESCAPE MOTOR

P_2 = PROBABILITY OF SUCCESS OF TOWER JETTISON MOTOR

P_3 = PROBABILITY OF SUCCESS OF PITCH CONTROL MOTOR

$$= \left[P_{EBW}^2 + 2 P_{EBW} (1 - P_{EBW})\right] \left[P_{PYROGEN\ IGNITER}\right] \left[P_{PROPELLANT}\right] \left[P_{LINER}\right] \left[P_{CASE}\right] \left[P_{NOZZLE}\right]$$

$$P_4 = \left\{P_{TOWER\ STRUCTURE}\right\} \left\{P_{BOLT}^4 + 4P_{BOLT}^3 (1 - P_{BOLT})\right\}$$

Figure 27. Launch Abort and Tower Jettison Reliability Logic Diagram

motor, and the resulting data will be compared with post fire measurements to determine the safety margin. Elastromeric materials will be measured prior to installation to determine prefire thickness. Chamber liner thickness will be measured circumferentially and axially at several stations before and after firing to ascertain uniformity and thickness. These data will be correlated with liner weight in an attempt to eliminate some measurements in favor of spot checks on later motors. Aft closure prefire and postfire thickness will be measured to check the safety margin.

To monitor chamber temperature, thermocouples will be placed at 12 locations at propellant web stations along the motor chamber. Other thermocouples will be located on the chamber, chamber dome, aft closure, and nozzle to measure any temperature use.

Table 36. Launch Escape Motor State-of-the-Art and Required Reliability

Component	Reliability - Allowed Failures/10^6 Motors	
	Required	Predicted State of the Art
Exploding bridge wire (2)*	1000	1000
Pyrogen igniter	500	2000
Propellant	500	1500
Liner	100	950
Case	100	950
Nozzle (4)	800	1600
Total motor	2000	7000
*Not included in total motor		

Pitch Control Motor

The pitch control motor provides side thrust to establish command module trajectory. Table 37 indicates the apportioned and state of the art reliability values for the pitch control motor.

Table 37. Pitch Control Motor State-of-the-Art and Required Reliability

Component	Reliability - Allowed Failure/10^6 Motors	
	Required	Predicted State of the Art
Exploding bridge wire (2)*	1000	1000
Pyrogen igniter	10	25
Propellant	500	10
Liner	100	10
Case	100	10
Nozzle	100	145
Total Motor	810	200
*Not included in total motor		

The pitch control motor development program is proceeding on schedule. Major accomplishments were (1) delivery to S&ID of dummy motor No. 1 to be used in the mock-up of the launch escape system, (2) fabrication was begun and completed on the first development motors, (3) first development motors were successfully cast and processed on schedule, and (4) succesful static test firings of four development motors were completed.

Reliability studies and evaluations are in process to determine margin of safety and motor integrity. These studies are essentially the same as those described for the launch escape motor.

Review of decisions and corrective actions from the Material Review Board indicates that safety margins are being maintained and that corrective action is adequate to prevent reliability downgrading during production.

Tower Jettison Motor

The tower jettison motor provides the thrust to separate the tower structure from the command module. Required and state of the art reliability levels for motor components are indicated in Table 38.

Table 38. Tower Jettison Motor State-of-the-Art and Required Reliability

Component	Reliability - Allowed Failures/10^6 Motors	
	Required	Predicted State of the Art
Exploding bridge wire (2)*	1000	1000
Pyrogen igniters	3	25
Case	1	10
Insulation	1	10
Propellant	1	10
Fixed Nozzles (2)	43	145
Total Motor	50	200
*Not included in total motor		

Projected reliability plans provide for analysis of data accumulated from test firings of motors. The results of these analyses will furnish information as to (1) reliability goals attained, (2) establishment of new reliability criteria if required, and (3) general reliability effort during the next report period.

Reliability effort is directed toward classifying the environmental modes into critical, major, and minor effects so that the most severe environmental effects can be studied first. The self-induced environments associated with solid rocket firing will be included in the classification.

The tower jettison motor development program is progressing satisfactorily. A schedule slippage occured because of difficulties encountered in motor casting, but the problem has been resolved.

The most significant of the many accomplishments during this report period were as follows:

1. Pryogen development testing was initiated with four successful static tests using conventional squibs.

2. The first inert TE-380 motor, BP-9, was delivered to S&ID on 9 October 1962.

3. The electrical interface fixtures and supports specified by S&ID have been incorporated into the interstage structure design.

4. The first attempt to case development motors on 22 October 1962. was unsuccessful because of processing and casting problems.

5. Test bay facilities have been completed for motor static tests.

6. Twenty-four development motor cases have been received at Thiokol Corporation.

TEST PROGRAM

Launch Escape Motor and Pitch Control Motor

Lockheed Propulsion Company conducted two successful development motor firings, during which lockheed and S&ID performed acoustical field measurements and recorded temperatures and vibration and shock levels. Data from these tests are currently being analyzed. Preliminary examination reveals that the near-field acoustic levels were approximately 167 db maximum with peaks at 2000 cps. The ignition shock level appears to be of significant magnitude, but the exact level is not presently available.

The following test activities are planned during the next reporting period: (1) continued launch escape motor development testing, (2) commencement of pitch control motor development tests, (3) completion of the igniter development program, and (4) resubmittal of the test plan.

Tower Jettison Motor

Thiokol Corporation successfully fired its first tower jettison motor as part of the development test program during this quarter. All test objectives were satisfactorily accomplished.

The next quarter will be devoted to a continuation of the planned development test program highlighted by simulated altitude firing at Arnold Engineering Development Center, Tullahoma, Tennessee.

Effects of Qualification Program Reorientation

No changes have resulted from the recent qualification program redirection.

Table 39. Launch Escape Motor, Pitch Control Motor, and Tower Jettison Motor

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Ignition EBW			Provides ignition of B-kno_3 pellets	Open-bridge wire	See note 4.	Fail to perform required function	None, redundant EBW	None	None
B-kno_3 pyrogen			Provides ignition of pyrogen	Pellets fail to ignite due to chemical decomposition		Motor ignition will not occur	See note 1.	See note 2.	None
Pyrogen			Provides ignition of propellant	Fail to ignite due to resin-rich surfaces or surface leaching		Motor ignition will not occur	See note 1.	See note 2.	None
				Overignition-pyrogen provides too much gas		Cracks, fissures, or break-up	See note 1.	See note 2.	None
Propellant			Provides motor thrust	Cracks, pits, or voids		Uneven burning, non-reproducible burn time	See note 3.	See note 2.	None
				Bond failures (thermal elongation grain-liner bond)		Poor ballistic reproducibility	See note 3.	See note 2.	None
Pressure vessel			Contains propellant and becomes pressure vessel after ignition	Maximum normal pressure case rupture		Catastrophic	See note 1.	See note 2.	None
Forward closure (launch escape motor only)			Provides forward containment of pressure vessel	Ruptures or blows off	See note 4.	Catastrophic	See note 1.	See note 2.	None
Aft closure and nozzle assembly			Provides aft containment of pressure vessel	Ruptures, cracks, or blows off		Catastrophic	See Note 1.	See note 2.	None

Table 39. Launch Escape Motor, Pitch Control Motor, and Tower Jettison Motor (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Action
						Component Performance	Subsystems Performance	Apollo	
Nozzle, insert			Liner to protect metal from high temperature	Cracks, excessive erosion		Thrust vector will change	Negligible	None	None
closure			Protects nozzle until firing	Pressure leaks		Negligible	None	None	None
expansion cone			Provides thrust effect	Cracks or voids		Thrust vector will change	Negligible	None	None

NOTES:
1. Should boost failure occur, tower jettison and launch escape motor must function for crew safety and mission success. For normal mission, launch escape motor is redundant to tower jettison motor.
2. Mission success and crew safety are both lost if tower jettison motor fails after launch escape motor has fired because of boost failure.
3. Providing premature burn-out does not occur, sufficient thrust will be furnished to provide mission success and crew safety.
4. All motor firing will occur during mission phases 1, 2, and 3.
5. All motors are of comparable design, thus failures are of the same order and type.

[redacted]

[redacted]

MECHANICAL SUBSYSTEM

During the past quarter, comparative evaluations were made on various mechanical systems.

Four release and thrust mechanisms applicable to the forward heat shield were compared. Reliability recommended a toggle holding mechanism released by a gas-generating cable cutter. The generated gas would then be used to power the thrusters. The recommended system meets the requirement of 0.999995.

Nine methods applicable to the forward heat shield, launch escape tower, and command module-to-service module separation systems were evaluated. Since all the methods evaluated could meet the system requirement of 0.999995, a qualitative comparison was made. A toggle latch and sear-type toggle were both rated 1 to 2 with a dual-initiated explosive bolt rated number 3.

Four proposed explosive bolt configurations were subsequently evaluated. All met the system requirement and were qualitatively compared. The recommended configuration contained a single main charge with dual initiation consisting of two primary charges, two bridgewires, and two electrical connectors.

CONFIGURATION ANALYSIS

Forward Heat Shield Separation System

A reliability evaluation of four release mechanisms applicable to the forward heat shield separation system was made. One of the release mechanisms studied was a sliding collet-type latch release. Initiation of an explosive charge by means of a low-energy detonating cord fails the shear pins. This allows the energy of the explosive charge to drive the collet retaining member to the unlocked position, which permits release of the latch. This release mechanism has a reliability of 0.99984 based mainly upon the loose parts that make up the release. The system appears to be very susceptible to jamming. Installation and accessibility are also problems, and this system cannot be checked out. There are several explosive interfaces, and for separation of the heat shield to occur, all four collet-type latch releases would have to function.

The three remaining release mechanisms are all very similar and are designated as (1) toggle release mechanism gas generator, (2) toggle release mechanism electrical firing, and (3) toggle release mechanism detonator manifold.

The toggle release mechanism gas generator system consists of two gas-generating cable cutters, each of which cuts both cables and actuates two of the four thrusters. The system is redundant, since two out of four thrusters will separate the heat shield. Each cable cutter releases the tension in both cables and actuates all four toggle release mechanism. Thus, there are four toggle releases and four thrusters. The toggle release mechanism has been used very successfully by NAA as a bomb and drop-tank release mechanisms on military aircraft. The toggle release system is simple to install anytime before the launch escape tower is placed on the command module. This system has a reliability of 0.999995. A problem that could result with this system might occur if one or both cutters did not completely cut the cables locking the system. This is not likely to happen, but can be overcome by using a large enough charge for each cable cutter. The tubing from the gas generator would be difficult to damage because of its dimensions and could even be formed into the honeycomb so damage would be almost impossible. The fairing that enclosed the cable is the most reliable means for ensuring safe operation. The fairing could be omitted and pulleys could be used, but this would require tie-downs and more pieces of equipment with no further increase in reliability.

The toggle release mechanism systems that were evaluated are very similar. One configuration uses an electrical firing system to ignite an explosive charge in each thruster; another uses a detonator manifold with an associated low-energy detonating cord to ignite an explosive charge in each thruster, and the remaining one uses a gas generator with associated tubing that generates the power for the thrusters. The toggle release mechanism with electrical firing and detonator manifold have a reliability of 0.999944 and 0.999935 respectively. The reliability numbers, block diagrams, features, and problem areas of each of the four systems are shown in Figures 28 through 31.

It has been determined that the toggle release mechanism system with gas generator is more reliable than the sliding collet-type latch release system or the toggle release mechanism with electrical firing or detonator manifold. This conclusion is based upon reliability, design principles, and chances for human error.

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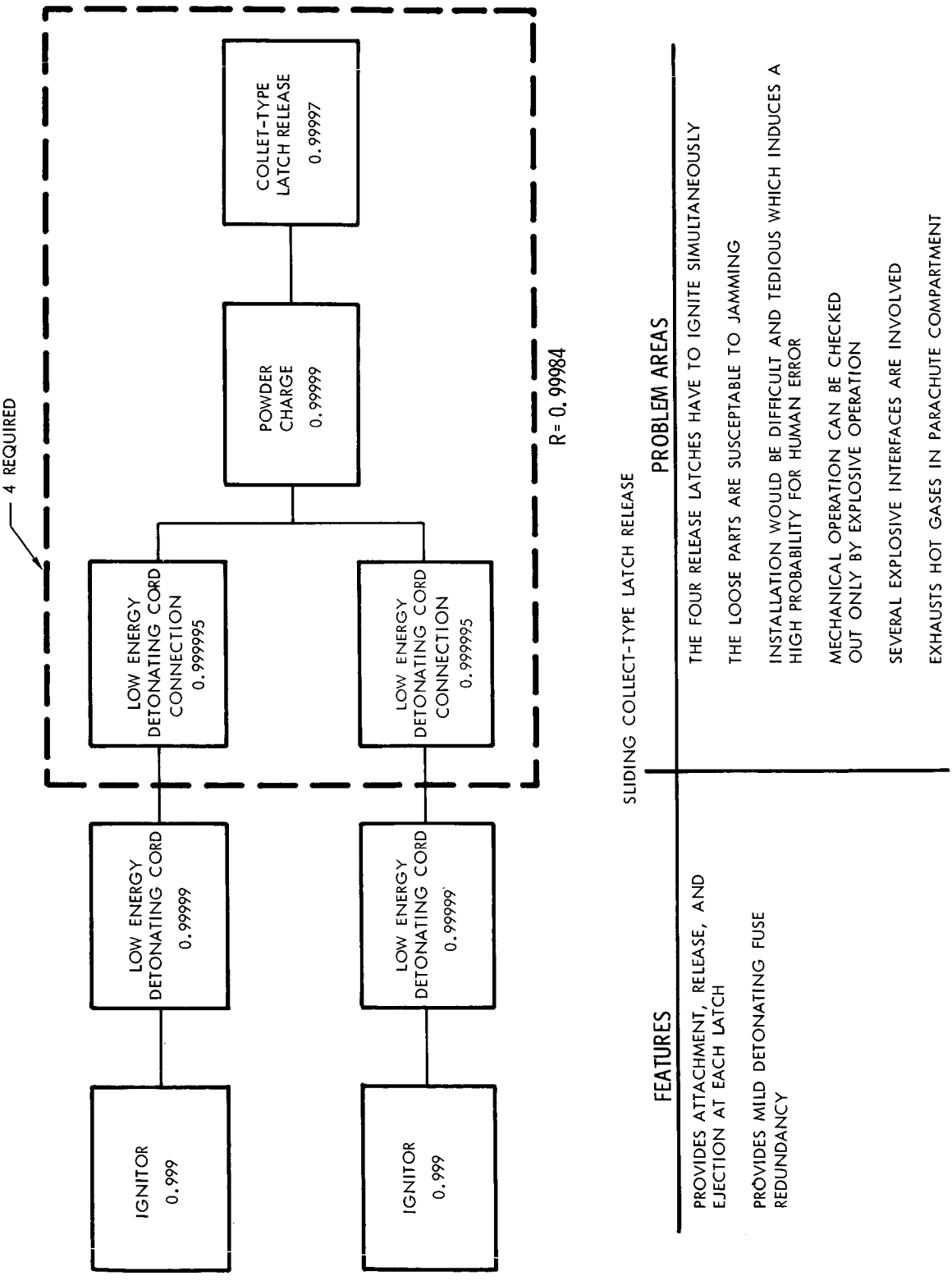
4 REQUIRED
COLLET-TYPE LATCH RELEASE 0.99997
POWDER CHARGE 0.99999
LOW ENERGY DETONATING CORD CONNECTION 0.999995
LOW ENERGY DETONATING CORD CONNECTION 0.999995
LOW ENERGY DETONATING CORD 0.99999
LOW ENERGY DETONATING CORD 0.99999
IGNITOR 0.999
IGNITOR 0.999
R = 0.99984
SLIDING COLLECT-TYPE LATCH RELEASE
FEATURES
PROVIDES ATTACHMENT, RELEASE, AND EJECTION AT EACH LATCH
PROVIDES MILD DETONATING FUSE REDUNDANCY
PROBLEM AREAS
THE FOUR RELEASE LATCHES HAVE TO IGNITE SIMULTANEOUSLY
THE LOOSE PARTS ARE SUSCEPTABLE TO JAMMING
INSTALLATION WOULD BE DIFFICULT AND TEDIOUS WHICH INDUCES A HIGH PROBABILITY FOR HUMAN ERROR
MECHANICAL OPERATION CAN BE CHECKED OUT ONLY BY EXPLOSIVE OPERATION
SEVERAL EXPLOSIVE INTERFACES ARE INVOLVED
EXHAUSTS HOT GASES IN PARACHUTE COMPARTMENT

Figure 28. Sliding Collet-Type Latch Release

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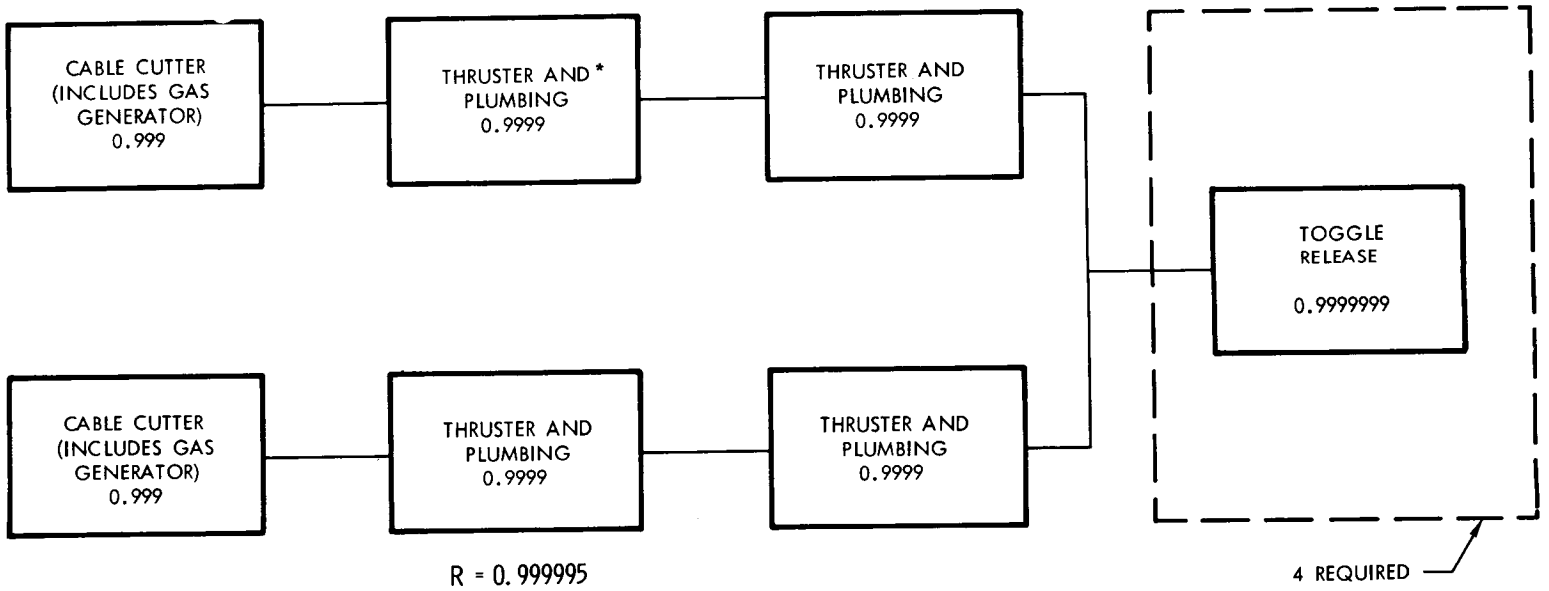
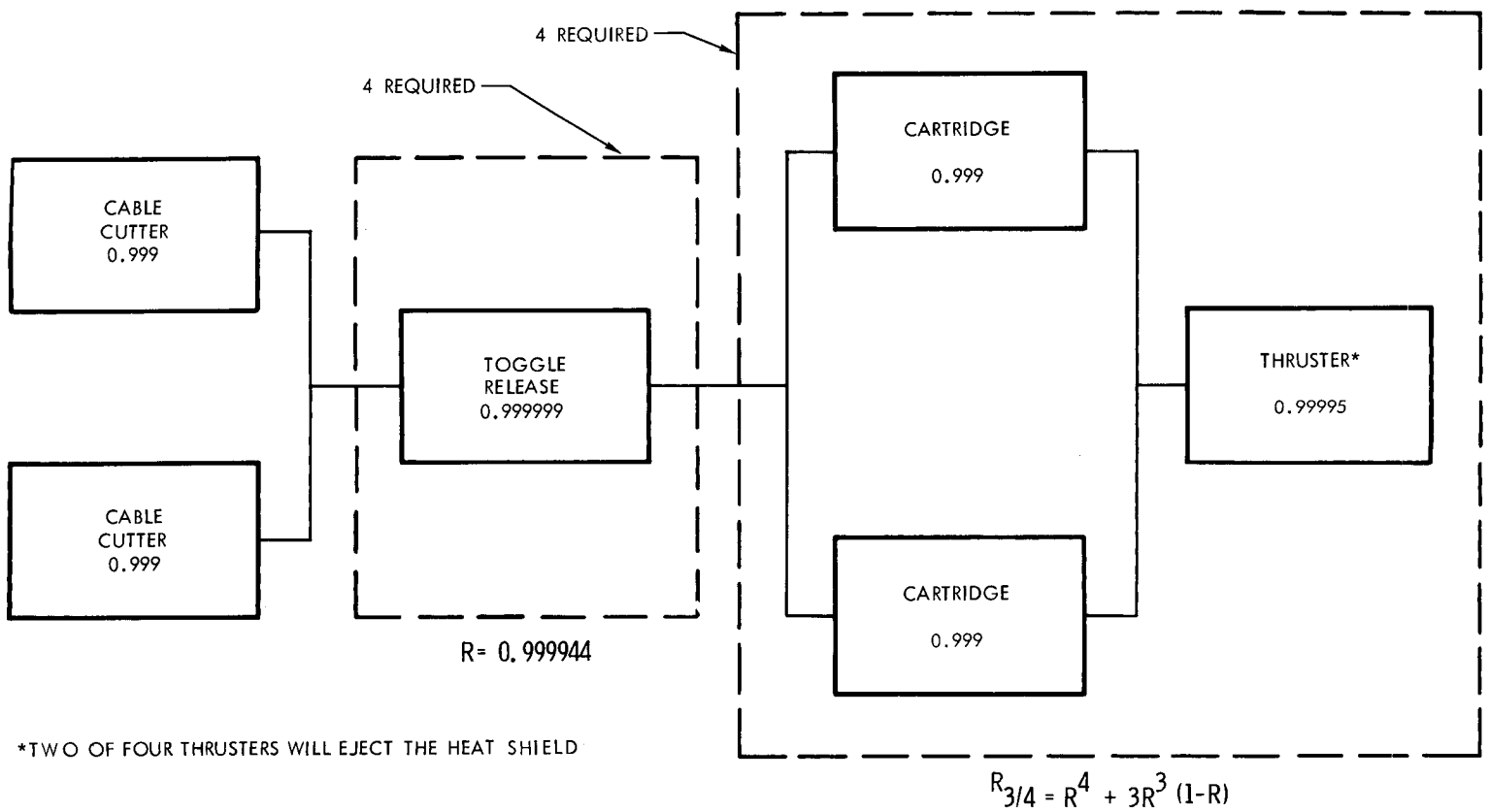

CABLE CUTTER (INCLUDES GAS GENERATOR) 0.999
THRUSTER AND * PLUMBING 0.9999
THRUSTER AND PLUMBING 0.9999
CABLE CUTTER (INCLUDES GAS GENERATOR) 0.999
THRUSTER AND PLUMBING 0.9999
THRUSTER AND PLUMBING 0.9999
TOGGLE RELEASE 0.9999999
R = 0.999995
4 REQUIRED
*TWO OF FOUR THRUSTERS WILL EJECT THE HEAT SHIELD
$R_{3/4} = R^4 + 3R^3 (1-R)$

FEATURES	PROBLEM AREAS
PROVIDES REDUNDANT SYSTEMS	POSSIBILITY OF JAMMING GUILLOTINE IN CABLE (REMOTE) EXISTS
PROVIDES MECHANICAL PRINCIPLE WHICH HAS BEEN PROVEN (HIGH CONFIDENCE LEVEL)	
PROVIDES SINGLE EXPLOSIVE FUNCTION	
PROVIDES RELIABILITY	
LOAD PATHS ARE MORE IDEALLY LOCATED FOR STRUCTURE	

Figure 29. Toggle Release Mechanism - Gas Generator



FEATURES	PROBLEM AREAS
ANY THREE THRUSTERS WILL EJECT	ACCESSIBILITY
ELECTRICAL INITIATION AT EACH ACTUATOR	SEVERAL EXPLOSIVE INTERFACES ARE INVOLVED
PROVIDES RELIABLE MECHANICAL RELEASE	

Figure 30. Toggle Release Mechanism - Electrical Firing

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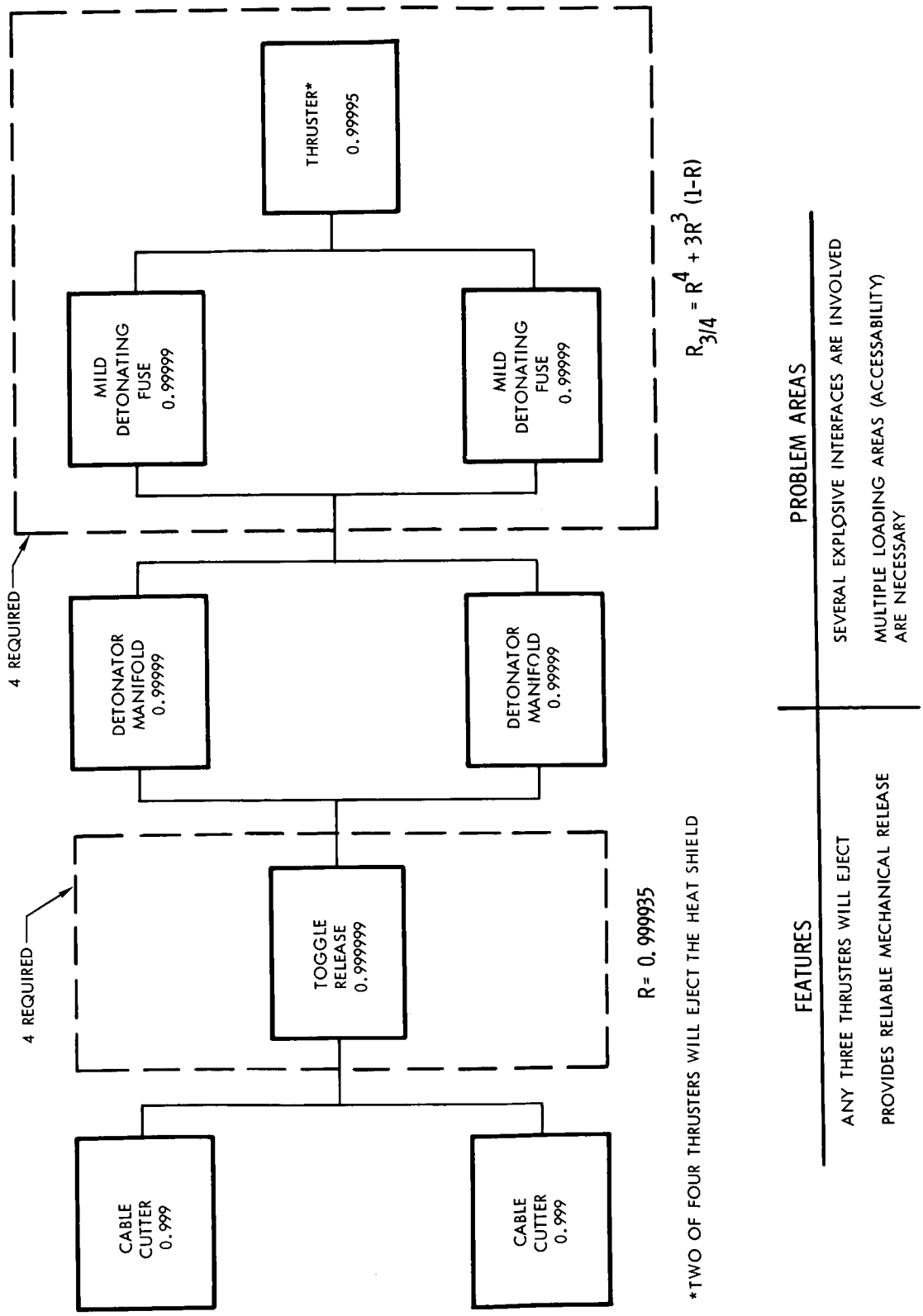

Figure 31. Toggle Release Mechanism - Detonator Manifold

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Separation Methods

A reliability evaluation was made of nine separation methods applicable to removal of the forward heat shield, launch escape tower separation and command module-service module separation. Numerically, each of the nine methods meet the reliability requirement, so a comparative qualitative evaluation was made. At the time the evaluation was made, weights were not available and, therefore, were not considered as a factor in the order of preference.

The factors considered in the evaluation include the possible means of signal distribution, the number of electrical pyrotechnic interfaces, the checkout capability, simplicity of design and operation, and system requirements for fragmentation protection. Table 40 lists the systems evaluated, the factors considered in the evaluation, and an order of preference.

Table 41 lists the advantages and disadvantages of the three methods of signal distribution that could be used, electrical, mechanical, and pyrotechnic.

Explosive Bolt Configurations

A reliability evaluation of four proposed explosive-bolt configurations was made. Numerically, each of the four configurations meets the reliability requirements, so a comparative qualitative evaluation was made. Figure 32 lists the configurations evaluated, the factors considered in the evaluation, and an order of preference.

PLANNED ACTIVITIES

Since the evaluations were made from preliminary drawings, planned activities consist of further analyses of future design drawings and studies of possible modes of any mechanical system.

Table 40. Separation Subsystem Evaluation

Separation Method	Possible Signal Distribution	Electrical - Pyrotechnic Interfaces	Checkout	Simplicity	Fragmentation Protection Required	Preference
1. Toggle latch	Mechanical	2	Yes	Good	No	1-2
2. Explosive bolt dual break	Electrical Pyrotechnic	8/6* 2	Yes No	Good	Yes	3
3. Dual linear shape shape charge to cut support	Electrical Pyrotechnic	8/6 2	Yes No	Fair	Yes	5
4. Existing toggle and explosive bolt	Mechanical-electrical Mechanical-pyrotechnic	2+8/6** 2+2	Yes No	Fair	Yes	8-9
5. Explosive bolt and explosive nut	Electrical Pyrotechnic	8/6 2	Yes No	Fair	Yes	4
6. Linear shape charge and explosive nut	Electrical Pyrotechnic	8/6 2	Yes No	Fair	Yes	6-7
7. Linear shape charge and explosive bolt	Electrical Pyrotechnic	8/6 2	Yes No	Fair	Yes	6-7
8. Explosive nut on toggle latch	Mechanical-electrical Mechanical-pyrotechnic	2+8/6 2	Yes No	Fair	Yes	8-9
9. Sear-type toggle	Mechanical Electrical Pyrotechnic	2 0 2	Yes Yes No	Good	No	1-2

* 8 interfaces for the launch escape tower or forward heat shield separation system; 6 interfaces for the command module - service module separation system

** 2 interfaces for the mechanical portion; 8 or 6 for the electrical depending upon the system being considered

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Table 41. Signal Distribution Evaluation

Electrical	Mechanical	Pyrotechnic
Advantages		
1. Light weight 2. Ease of checkout 3. Flexibility of installation	1. Simplicity 2. Single electrical-mechanical interface 3. Proven mechanical principle 4. Minimum susceptibility to inadvertent firing 5. Can be checked out	1. Capability of high reliability through redundancy 2. Single electrical-mechanical interface
Disadvantages		
1. Multiple electrical-pyrotechnic interfaces 2. Increased probability of inadvertent firing	1. Cold welding 2. Weight	1. Multiple pyrotechnic interfaces 2. Exhaust gases 3. Multiple primary explosives 4. Cannot be checked out

EXPLOSIVE BOLT CONFIGURATION EVALUATION

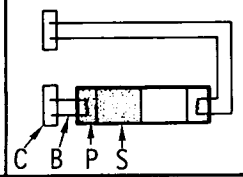
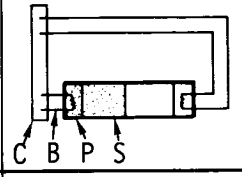
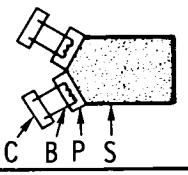
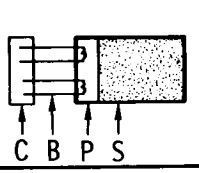
CONSIDERED CHARACTERISTIC	CONFIGURATION			
				
SECONDARY EXPLOSIVE (MAIN CHANGE) (S)	REDUNDANT	REDUNDANT	SINGLE	SINGLE
PRIMARY EXPLOSIVE (P)	REDUNDANT	REDUNDANT	REDUNDANT	SINGLE
BRIDGE WIRE (B)	REDUNDANT	REDUNDANT	REDUNDANT	REDUNDANT
ELECTRICAL CONNECTOR (C)	REDUNDANT	SINGLE	REDUNDANT	SINGLE
NO. OF SINGLE FAILURES CATASTROPHIC	0	1	1	3
PROBABILITY OF OCCURENCE	-	MEDIUM	LOW	HIGH
RELATIVE COMPLEXITY	HIGH	HIGH	MEDIUM	LOW
RELATIVE WEIGHT	HIGHEST	HIGH	MEDIUM	LOW
CAN STANDARD ELECT-PYROTECHNIC HEADER BE USED?	NO	NO	YES	YES
MUST BOLT INSTALLATION BE MADE ON PAD?	YES	YES	NO	NO
CAN BRIDGEWIRE BE CHECKED FOR CONTINUITY AWAY FROM SECONDARY EXPLOSIVE?	NO	NO	YES	YES
RELIABILITY RATING	2	4	1	3

Figure 32. Explosive Bolt Configuration Evaluation

[redacted]

EARTH LANDING SUBSYSTEM

Accomplishments on the earth landing subsystem during the past quarter include further investigation into the use of the one drogue configuration and performance of a failure effect analysis of the subsystem. From the failure effect analysis, the reliability of the parachute recovery portion of the earth landing system is estimated to be 0.99997. This compares favorably with the requirement of 0.99996. The mathematical equations and the results of the failure effects which formed the basis for this analysis are presented herein. The failure effects analysis is shown in Table 44 at the end of this section.

CONFIGURATION ANALYSIS

Subsystem Description

The primary purpose of the earth landing subsystem is to provide safe landing of the Apollo command module under reentry or abort conditions. At the present time, the subsystem consists of 1 drogue, 3 pilots, 3 main chutes, a sequencer, and location aids.

The apportioned reliability of 0.99994 will be met, using present state-of-the-art reliability. The ground rule under which this system is evaluated is successful operation of the single drogue and two out of three main chutes, or, if the single drogue fails, successful operation of all main chutes.

The validity of this ground rule depends on the capability of the main chutes to withstand the loads imposed on them if the drogue fails to work. The main chutes are designed to 64 q and tested to 96 q. Terminal q of the command module, aft heat shield forward, is 87 q. Work is being done at the present time to establish the reliability of the main chutes at these higher q numbers. Recent development tests have had some successful operation up to 106 q.

Under this ground rule, the problem of the apex forward trim position still exists. The command module being stable apex forward from a high altitude abort is experiencing a 140 q. In the case where the drogue fails, the main chutes would never survive this high q number. A possible solution would be to redesign the main chutes to a higher q number or to eliminate the apex forward trim condition. Presented herein are reliability equations for parachute deployment and a failure effect analysis.

Parachute Portion of Earth Landing Subsystem Equations

These equations for parachute deployment will cover, not only mission success, but crew safety, since they are the same for a normal earth landing. These equations are based on the ground rules previously stated and exclude manual override capability and abort conditions.

$$R_{\text{parachute deployment}} = R_{\text{drogue}} R_{\text{sequencer}} R_{\text{2 out of 3 mains}}$$

$$+ (1 - R_{\text{drogue}}) R_{\text{sequencer}} R^3_{\text{3 out of 3 mains}}$$

$$R_{\text{drogue}} = \left[1 - \left(1 - R^2_{\text{baro switch}} R_{\text{relay}} R_{\text{squib}}\right)^2\right] R_{\text{mortar}} R_{\text{drogue}}$$

$$R_{\text{sequencer}} = \left[1 - \left(1 - R_{\text{time delay}} R^2_{\text{baro switch}} R_{\text{relay}}\right)^2\right]$$

$$R_{\text{main}} = \left[1 - \left(1 - R_{\text{squib}}\right)^2\right] R_{\text{mortar}} R_{\text{chute}}$$

$$R_{\text{2 out of 3 mains}} = R^3_{\text{main}} + 3R^2_{\text{main}} (1 - R_{\text{main}})$$

$$R_{\text{parachute deployment}} = 0.99997$$

Table 42. Earth Landing Component Failure Rates and Reliability Values

Item	Failure Rate (million)	Reliability
Relay	6	0.999994
Time delay	30	.999970
Baroswitch	100	0.999900
Mortar	100	0.9999
Parachute	2000	0.998000
Hot wire squib	1000	0.999

TEST PROGRAM

Three parachute development drops were conducted during October, five during November, and (at time of writing) one during December. These drops were to verify structural integrity and high-speed capability, provide loads data, and evaluate cluster performance.

Laboratory tests, including functional and environmental testing of parachute fabric, were continued. Functional tests were conducted on pilot and drogue chute mortars, and firing tests were conducted for verification of velocity and deployment capabilities. Design of the sequencing system is in progress.

The procurement specification was revised to reflect the redirected qualification test program, (Specification MC 901-0001B).

Effect of Redirection of Qualification Test Programs

The numbers of test specimens have been reduced for qualification of the drogue and main chute disconnects, the drogue and pilot chute mortars, the sequence controller, and parachute fabric samples. The reliability demonstration requirements and statistical analysis of test results have been deleted. Test program responsibility has been assumed by S&ID. The qualification test sequence and detail test procedure have been revised to achieve a more realistic test program. The reduction in the numbers of specimens is as follows:

Table 43. Earth Landing System Test Specimen Reduction

Component	Number of Specimens	
	Original	Revised
Disconnect, drogue chute	200	50
Disconnect, main chute	200	50
Mortar, drogue chute	6	4
Mortar, pilot chute	6	4
Sequence controller	8	4
Parachute fabric samples	Undefined	40

Table 44. Earth Landing Subsystem Sequencer Failure Effects Analysis

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Error
						Component Performance	Subsystems Performance	Apollo	
Battery			Supplies power to one channel of sequencer	Fails to supply current to one channel of sequencer		Channel fails	Loss of redundancy	Loss of redundancy	No
				Supplies less than rated current to one channel of sequencer					
Forward heat shield baro switches (in series)			Senses altitude to release forward heat shield	Both switches in closed position		As soon as system is armed current is passed through baro	Forward heat shield released when system armed	Loss	No
				Both switches grounded One switch grounded, other normal operation		No effect			
Forward heat shield relay			Activities Squib	Fails to operate		Channel fails	Loss of redundancy	Loss of redundancy	No
Drogue time relay			Used on low altitude abort to deploy drogue	Fails to operate		Channel fails	Loss of redundancy	Loss of redundancy	No
				Operates prematurely		Normal earth landing no effect because of baro switches	No effect	No effect	No
						Low altitude abort, current passes through time delay	Drogue deploys at same time heat shield releases	Possible loss due to fouling of drogue chute	No
Drogue baro switches			Senses altitude to deploy drogue	Both switches in closed position		As soon as system is armed current is passed through baro	Heat shield released and drogue deployed simultaneously	Chute	No

Table 44. Earth Landing Subsystem Sequencer Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Error
						Component Performance	Subsystems Performance	Apollo	
Drogue baro switches (Cont)				Both switches grounded One switch grounded other normal operation		Channel fails	Loss of redundancy	Loss of redundancy	
				One switch closed position other normal operation		No effect			
Drogue relay			Activates Squib	Fails to operate		Channel fails	Loss of redundancy	Loss of redundancy	No
Pilot chute time delay			Used on low altitude abort to deploy pilot chutes	Fails to operate		Channel fails	Loss of redundancy	Loss of redundancy	No
				Operates prematurely		Normal earth landing No effect because of baro switches	No effect	No effect	No
						Low altitude abort current passes through time delay	Pilot deploy at same time heat shield releases	Possible loss due to fouling of main chutes	
Pilot chute baro switches			Senses altitude to deploy pilot chutes	Both switches in closed position		As soon as system is armed current is passed through baro	Pilot chute deployed and heat shield released simultaneously		Nc
				One switch grounded other normal operation		Channel fails	Loss of redundancy	Loss of redundancy	
				One switch closed position other normal operation		No effect			
Pilct chute relay			Activates squib	Fails to operate		Channel fails	Loss of redundancy	Loss of redundancy	No

Table 44. Earth Landing Subsystem Sequencer Failure Effects Analysis (Continued)

Item	Drawing No.	Electrical Reference Designation	Function	Failure Type	Mission Phase	Failure Effect			Possible Crew Error
						Component Performance	Subsystems Performance	Apollo	
Impact switch time delay			Time delay so opening shock of parachutes won't set off impact switch	Fails to operate		Channel fails	Loss of redundancy	Loss of redundancy	No
				Operates prematurely		Signal is passed thru time delay simultaneously with deployment of pilot chutes	Possibility of releasing main chutes if impact switch is activated by opening shock of parachutes and pilot has activated his manual switch		
Impact switch			Used in combination with manual switch to release main chutes after reentry and touchdown	Operates prematurely		Signal is passed thru impact switch simultaneously with deployment of pilot chute		Loss	No
				Fails to operate		Channel fails	Loss of redundancy	Loss of redundancy	No

PLANNED ACTIVITIES

Future activities planned for analysis of this subsystem include investigation into the possibility that, if the drogue and one main fail, the remaining two main chutes will be sufficient to land the command module safely and a more detailed failure effect analysis as the design is firmed up. The subcontractor has been requested to conduct a study of the reliability of the main chute cluster at 96 q. They have also been requested to determine the optimum number of reefing line cutters to minimize the difference in disreefing time.

ELECTRONICS SUBSYSTEMS

This section contains a description of all electronics subsystems within the spacecraft and the various components within the subsystems. The reliability estimates for these individual components and subsystems are of secondary importance; the total spacecraft reliability, described in a later section that includes the backup of components within one subsystem by components in another, gives a more important evaluation of the subsystems.

COMMUNICATIONS AND DATA SUBSYSTEM

During this period, analyses were performed to determine the status of the reliability of the communications and data subsystem when integrated with the other Apollo electronic subsystems, including a determination of spares necessary for meeting the requirements. A new reliability apportionment was established based on the lunar orbital rendezvous mission and the studies previously mentioned. The analyses indicate that the configuration is adequate to meet the reliability requirements with the use of additional spares. During the next period, these analyses will be refined to determine a more efficient sparing plan, based on replacing parts at a lower level than has presently been considered.

CONFIGURATION ANALYSIS

Subsystem Description

The communications and data subsystem contains that equipment which is used for communication between the spacecraft and the ground during all phases of the mission. This includes the transmission of technical data, which will be used for mission analysis, as well as voice communication. The VHF/AM transmitter-receiver equipment provides voice communication between the spacecraft and earth during the near-earth phases (up to 4000 miles). Beyond this range, the Deep Space Information Facility (DSIF) equipment is used for voice communication. Associated with this equipment are three audio centers, one for each crew member, which provide the link between the crew member and the transmission-reception equipment. The VHF/2kmc omni-directional equipment is utilized with the DSIF and VHF equipment in all phases except earth landing. Associated with the antenna are the multiplexing and switching units. The 2kmc omni-direction portion of this antenna provides backup for the 2kmc high-gain antenna. For the processing of sensory data, the premodulation processor equipment, signal conditioning equipment, and pulse code modulation telemetry equipment are provided. The premodulation processor equipment also performs the switching, gating, and mixing associated with the various modes of information transmission. The VHF/FM transmitter is used for transmission of telemetry information during the near-earth phases. The C-band transponder equipment is used in conjunction with the radar tracking facilities during the earth phases. During the recovery phases, the VHF recovery beacon provides beyond line-of-sight direction finding capability as well as voice communication. The data storage equipment is used to store telemetry data during critical time periods and when real time-data transmission is not possible.

Figure 33 depicts the configuration of the communications and data subsystem for mission success. The communications and data subsystem contributes to the probability of mission success by providing a secondary method for the crew members to determine the position of the spacecraft relative to its programmed position and the manual corrections necessary to achieve the programmed position. It also aids in the recovery of the spacecraft after earth landing.

Table 45 shows a compilation of the components of the communications and data subsystem, and estimated reliabilities with and without proposed redundancies, and apportioned reliabilities. The component failure rates were calculated, utilizing Minuteman (or equivalent) parts failure rates based upon nominal operation to 50 percent electrical stress under ground environments. The rationale for the use of ground environments failure rates is that the environmental control system aboard the Apollo will presumably maintain an environment comparable to ground environments. Risk factors accounted for are essentially those under the design control of system, equipment, circuit, and component part engineering. In this process it is assumed that sound manufacturing and quality control practices are involked by engineering through appropriate control measures.

The redundancy and spares considered for the direct mission and the first lunar orbital rendezvous mission are those presently contemplated for the communication and data subsystem.

The spares list for the second lunar orbital rendezvous mission analysis was modified by the addition of a spare signal conditioner and the deletion of the spare VHF/AM transceiver, which resulted in a net decrease of 3.6 pounds in the spares list for the communication and data subsystem. The resultant decrease of subsystem failures from 65.1 to 50.4 failures per 10,000 missions reflects a more optimum spares configuration for the subsystem.

The third lunar orbital rendezvous mission analysis was performed with a fourth reduction in spares weight by deletion of the spare VHF/AM transceiver and no additional sparing for other components. This resulted in a subsystem failure rate of 48.5 failures per 10,000 missions, which is a net decrease of 16.6 subsystem failures per 10,000 missions together with a net spares weight decrease of 13.6 pounds compared with the first lunar orbital rendezvous mission analysis; this reflects the increase in the reliability which is obtained through the use of Minuteman parts throughout the entire Apollo electronics.

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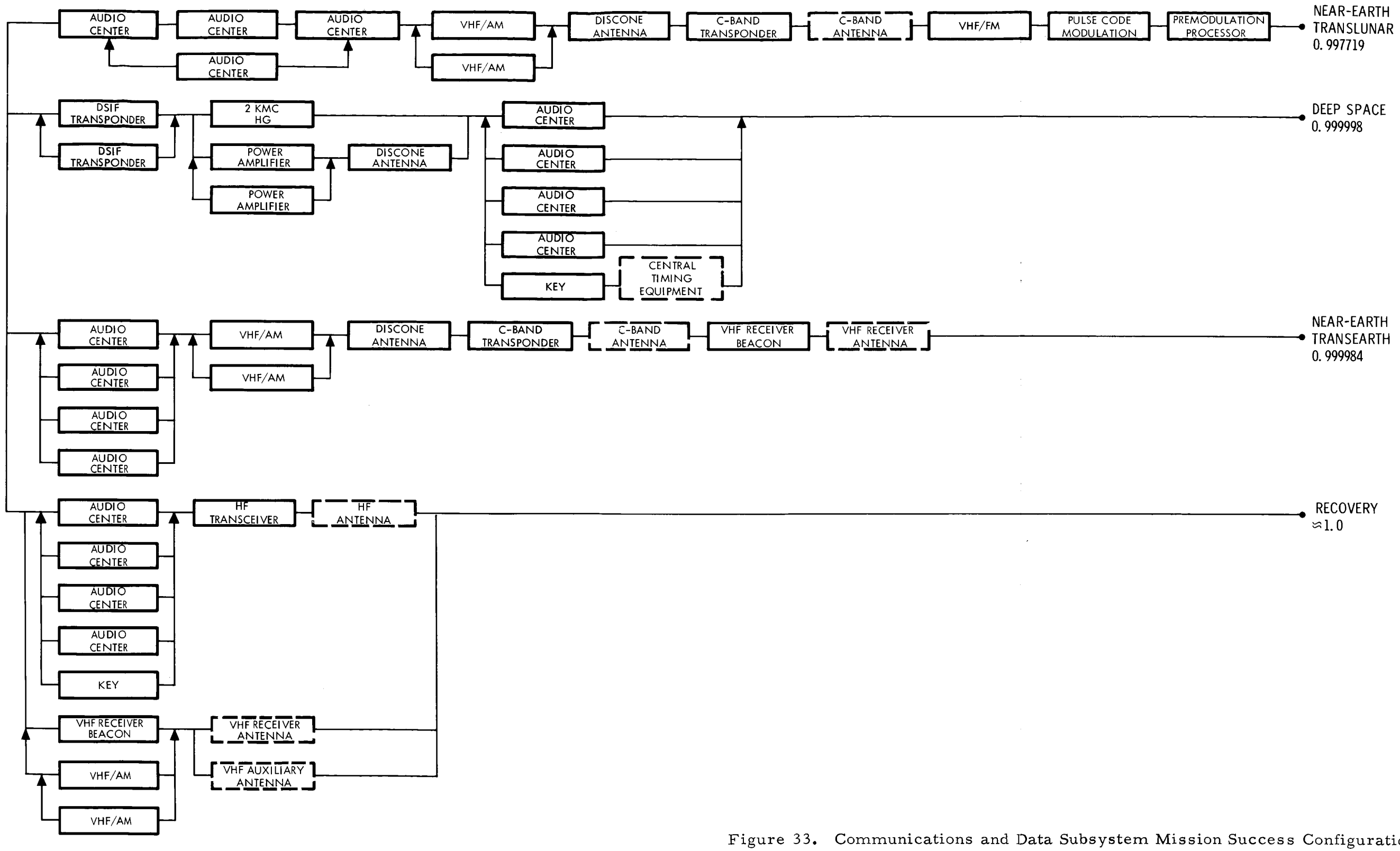


Figure 33. Communications and Data Subsystem Mission Success Configuration

Table 45. Communication and Data Subsystem Mission Reliability

Equipment	Direct Mission	Lunar Orbital Rendezvous Mission								
		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Minuteman	Spares	
	Minuteman		Number	Weight (lb)		Number	Weight		Number	Weight (lb)
VHF recovery beacon	0.999995	1.0			0.99940			0.99960		
HF transceiver	0.999992	1.0			1.0			1.0		
VHF/AMtransceiver	0.999954	1.0	1	13.6	0.99972			0.99985		
VHF/FM transmitter	0.999972	1.0			1.0			1.0		
Audio center	0.999449	1.0	1/4	1.6	0.99483	1/4	1.6	1.0	1/4	1.6
Data storage	0.999944	1.0			1.0			1.0		
C-band transponder	0.999939	0.99913			0.99976			0.99974		
DSIF transponder	0.99869	0.99913	1	16	0.99900	1	16	0.99935	1	16.0
DSIF power amplifier	0.999756	1.0	1	12.5	1.0	1	12.5	1.0	1	12.5
Controls and displays	0.999300	1.0			1.0			1.0		
Premodulation processor	0.999592	1.0			1.0			1.0		
Signal conditioner	0.999739	0.99522			0.99724	1	10	0.99673		
PCM telemetry	0.99810	1.0			1.0			0.99988		
VHF/2 KMC omniantenna	0.999999	1.0			1.0			1.0		
Total	0.99442	0.99349		43.7	0.99496		40.1	0.99515		30.1

TEST PROGRAM

Qualification Test Plan

The Collins Radio Company document AR-113-3 is to be rewritten and resubmitted, incorporating the latest qualification test philosophy. The hardware utilization list has been revised to reflect the requirements for the reoriented qualification program.

General Test Program Status

General ground rules for the environmental testing and qualification of Apollo equipment were established in the reoriented qualification test program. The program presented to NASA outlined minimum acceptable requirements. The salient features are presented in the following paragraphs.

The quantities of test specimens available for the test program have been reduced approximately 50 percent.

A maximum test effort will be expended to utilize available test time and to extract the maximum usable data from the test units available. This requires reuse of test units, refurnishment of test units as required, and extreme care in evaluating test data.

Considerable time and effort in coordination and study of the environmental levels have resulted in a general reduction of the test levels. Supplier necessity for additional facilities have been reduced as a result of this easing of requirements.

The following mission test profile (Mission Life Test) was developed.

1. Each test unit will be subjected to 400 hours of ground checkout time in an ambient environment.

2. Each test unit will be subjected to a simulated launch phase consisting of vibration and temperature.

3. Each test unit will be subjected to an environment of 100 percent oxygen at 5 psi for a minimum of 100 hours and a maximum of 336 hours, simulating translunar and trans-earth phases.

4. Equipment which must operate during and/or after reentry will be subjected to a reentry high-temperature profile.

5. Equipment which must operate for the recovery phase will be subjected to a shock environment simulating earth landing.

6. Succeeding simulated missions will consist of repeats of steps (1), (2), and (4). Step (3) will be conducted in room ambient environment.

7. The communications and data subsystem becomes qualified after completion by all test units of two mission simulations. After a total of four mission simulations, the equipment is operated to end-of-life in a room ambient, test-bench environment to acquire additional data on repeatability.

8. A shock test to provide assurance that the equipment mounting is adequate to eliminate any physical hazard to the astronaut upon earth landing will complete the testing.

SUBCONTRACTOR MANAGEMENT

On November 27 and 28, 1962, a meeting was held with Collins Radio Company in Cedar Rapids, Iowa, to discuss the effect of the use of high-reliability parts in the communications and data subsystem on the cost and weight of the equipment. This meeting was attended by representatives from the NAA/S&ID and NASA Reliability Office. The following is a summary of the items discussed and the conclusions reached as a result of this meeting.

1. Collins presently plans to use high-reliability parts for all communications and data equipment. Approximately 80 percent of these parts will be either Minuteman or other high-reliability parts that have already been developed. Parts developed to Minuteman standards, including those that are a complete new development for performance, will make up approximately 20 percent of the parts.

2. Use of high-reliability parts will raise the estimated reliability from approximately 26 percent to the value required to meet the requirement of 99.25 percent.

3. The total cost for the use of high-reliability parts, including serialization and traceability, will be approximately a 3-1/2 percent increase based on the total program cost.

4. The use of high-reliability parts will result in a negligible increase in the weight of the equipment over that used in Gemini. This increase is small, since approximately 50 percent of Minuteman parts are used in the Gemini equipment.

The major reason that the Apollo equipment weighs more than the Gemini equipment is the higher power output of the Apollo equipment and the types of construction used. As an example, the Gemini VHF/AM transmitter-receiver power output is three watts, while Apollo is five watts. Further, the Apollo equipment is constructed to carry structural loads and to provide a higher thermal conductivity than the Gemini equipment for lower operating temperatures of the parts.

5. Collins Radio Company stated that lowering the vibration requirements from 15 g's to 10 g's would have no effect on the construction of the equipment and would have no effect on the cost of environmental testing equipment.

From these discussions it was concluded that the use of high-reliability parts involves only a small increase in the cost and weight of the equipment but has a great effect on its reliability. Therefore, it is felt that Collins Radio Company should continue with its present plans to use these parts.

A similar meeting is planned with Minneapolis-Honeywell early in January 1963.

Detailed Analysis

The details of the studies presented by Collins are included below.

1. Parts usage—Collins has established the following plan for parts usage and development. The parts are to be procured to the following criteria in order of their ranking.

   a. Minuteman parts of the "B" level with serialization will be used where available.

   b. Parts are bought from Minuteman vendors using Minuteman production and testing procedures.

   c. Other industry high-reliability parts will be used.

   d. Parts that are being developed especially for the Apollo project, such as the r-f and i-f coils, will be developed to high-reliability standards. Detail specifications will be written to define these requirements.

The "B" level Minuteman parts are a level lower than the "M" level used in Minuteman production. They are manufactured and inspected to the same standards, but are not subjected to energized storage nor normally to serialization, whereas "M" level are. "B" level parts are packaged for a common carrier. (The energized storage consitutes storage by the equipment manufacturer for a period of three months with reduced voltage applied and periodic measurements. Serialization refers to serialization at the part level to allow for feedback of test data for determination of the probable cause of failures, the effect of manufacturing and process changes, and for subsequent configuration control and traceability. Packaging for a common carrier refers to standard packaging and handling techniques)

As an example of the expected distribution of parts, the VHF/AM transceiver will utilize the following percentages of parts. This distribution is considered typical for the total subsystem.

Parts	Percent
Minuteman parts	54
Parts developed to Minuteman standards	14
Other high-reliability parts	25
Newly developed parts	7

2. High-reliability parts— The following list contains the estimates made by Collins of the level of reliability which can be achieved using standard parts and using the program described in the preceding paragraph. In these calculations, the estimates were based on the actual operating times estimated by Collins.

Component	Standard Parts	High-Reliability Parts
HF transceiver	0.99865	0.999956
DSIF transponder	0.917	0.99135
DSIF PA	0.996	0.99974
C-Band	0.99728	0.99984
Intercom	0.925	0.9971
VHF/AM	0.983	0.9989
VHF/FM	0.9992	0.99994
PCM	0.316	0.955
	0.261	0.942

Collins proposed meeting the reliability requirement of 0.9925 by including redundant DSIF, VHF/AM transceiver, and PCM units. The redundancy is not consistent with S&ID provisions, which include only a redundant DSIF unit.

3. Cost of Minuteman parts — To provide an estimate of the cost differential that would be expected due to the use of Minuteman parts rather than Military standard parts, Collins Radio Company analyzed the VHF/AM transmitter-receiver. This unit was chosen since the units in Gemini and Apollo are very similar. The cost estimates were based on utilizing a screening technique during incoming inspection for the MIL standard parts as well as the Minuteman parts. This procedure is the one that was utilized for the Gemini equipment. It was estimated that the use of Gemini or Mercury parts with screening would reduce the $6,000 cost of the Apollo parts by approximately 25 percent.

   It was further estimated that eliminating serialization and traceability would reduce the cost of this unit by about 8 percent making a total reduction of 33 percent. This would result in approximately a 14-percent savings in the total cost of the unit, including manufacturing. Assuming that a comparable reduction could be effected on all parts,

the total reduction would be about 14 percent or a reduction of about 3-1/2 percent based on total program costs.

4. Comparison of Gemini and Apollo equipment—The VHF/AM transmitter for the Gemini and Apollo systems were compared to determine the reasons for the higher weight of the Apollo equipment. In most cases, the weight of the detail parts is nearly the same, since the Gemini equipment employs about 50-percent Minuteman parts and uses about the same voltage derating. In a few cases, parts have been added to the Apollo system to allow for additional derating, generally in the power stages. A major cause for the difference in the weight is the higher power output requirement for Apollo—five watts versus three watts. An additional increment of weight is due to the differences in construction technique. The Apollo modules are required to carry certain structural loads, while the Gemini equipment is not. Further, the Apollo structure is made heavier to provide better thermal conductivity and, thus, to allow the Apollo equipment to run cooler. In both cases the detail parts are mounted on metal boards.

5. Effect of reduced vibration environments—Collins was asked if the reduction in the amplitude of the vibration environment from 15 g's to 10 g's would affect the cost of the design of the equipment. It was stated that no change would be made in the design prior to testing. However, it would be expected that, with a lower level of vibration, there would be less chance of required redesign, which would tend to reduce the overall cost. It appears that the same type of vibration equipment would be required for either of these vibration levels.

6. Installation of parts—Collins could see no reason why the cost of installing high-reliability parts would vary from the cost of installing standard Military parts. They are presently designing their equipment so that the parts can either be welded or soldered. A decision will be made at a later date as to which procedure will be used after receiving an approval from S&ID and NASA. Collins stated that high-reliability parts could be welded just as easily as Military standard parts.

7. Traceability and Serialization—Collins plans to utilize serialization for all parts in communications and data equipment, except for resistors. An individual key punch card will be

utilized on all parts, thus providing traceability for all parts. The price quotes are based on this plan. It was estimated that the total cost for traceability and serialization would be approximately 20 percent of the direct labor assembly cost.

8. Welding Specification—Collins is presently developing welding techniques which will provide the basis for a specification to be published by 1 January 1963.

9. Conclusions and Recommendations—While the cost of high-reliability parts has an insignificant effect on the total program cost, the use of these parts is expected to have a significant effect on the component reliability. The incremental weight increase due to the use of Minuteman parts and additional stages of amplification is negligible. The major weight increase in the Apollo equipment is due to the additional power output and the weight of the packaging which must carry structural loads.

   It is recommended that the present program being pursued by Collins be maintained because of the extremely high increase in reliability and the relatively low increase in cost.

A "Data Storage Equipment Meeting" was attended at Collins Radio Company ISC, Newport Beach, California, by members of S&ID, Collins Radio Company, and Fairchild Camera and Instrument Company. Fairchild, Collins' choice of the data storage equipment bidders, presented a breadboard model of their tape deck. Under static conditions, their radical design, according to Collins and Fairchild, performs very closely to the required specifications. However, this model has never been tested under any but static laboratory conditions. This design of "iso-elastic drive," the name given to the friction belt-tape drive, is so new that there are no reliability test results available.

This simple drive design makes use of two separate capstans. Power is transmitted through the capstans to the belt and reels of tape with no slippage. Due to the separate and unequal velocities of the capstans and the slight elasticity of the belt, tension is maintained between the two reels of tape and a constant pressure is maintained on the tape head.

Fairchild gives a reliability estimate of 0.9999 for an operating time of 50 hours. This figure was composed almost entirely from the failure rates given in the Martin Handbook of Reliability Analysis and none from actual test, per verbal communications with D. Burdorf of Fairchild.

The justification of the Fairchild choice by Collins is approximately eighty-eight percent management status and twelve percent technical. The twelve percent technical is based mainly upon design, simplicity, and state-of-the-art. However, Collins' definition for state-of-the-art differs greatly from that of NAA. The scoring method used by Collins was weighted far too heavily on the management aspect; the normalization method employed by Collins misrepresents the spread of the bidding companies.

In summary, the Fairchild design is inadequate for the following reasons:

1. A model of the proposed design has not been built.

2. No reliability testing has been performed on the prototype.

3. No environmental testing has been performed on the prototype.

4. The nature of the drive mechanism of the recorder would make loading virtually impossible for an astronaut in a spacesuit because of the use of a perimeter belt drive.

5. Fairchild has had no previous experience in the recorder field. Therefore, if this type of design proved unworkable, the possibility of an acceptable alternate design by Fairchild would be remote.

The December monthly coordination meeting with Collins Radio Company was held at Cedar Rapids, Iowa, from 4 December through 7 December 1962. The purpose of the meeting was to clarify the recent revision to the **Statement of Work**, SID-62-12, the **General Specification**, MC 999-0023, and the associated equipment specifications for the communications and data subsystem. The reliability sections of the above documents were clarified to the satisfaction of both NAA/S&ID and Collins Radio.

PLANNED ACTIVITIES

During the next time period Collins Radio Company is expected to submit spares recommendations at the **module level rather than the component** level presently being considered. These data will be used to re-evaluate the requirements for spares necessary to meet the reliability requirements at the minimum weight level. The analyses discussed in this section will be further refined to establish a new reliability apportionment, which will then be issued to the subcontractor.

[redacted]

GUIDANCE AND NAVIGATION SUBSYSTEM

During this quarter, the effects of the component reliabilities, as estimated by Massachusetts Institute of Technology, on the reliability of the spacecraft have been evaluated and the spares necessary within this system to meet the spacecraft requirements have been evaluated. This study has also resulted in a preliminary subsystem reapportionment, based on the lunar orbital rendezvous mission. The analyses indicate that spares in addition to those presently planned are necessary for meeting the requirement. A further study indicates that, if high-reliability parts were used rather than those planned by Massachusetts Institute of Technology, the spares weight could be reduced. During the next quarter, these analyses will be refined to better define spares requirements and to finalize the reapportionment for the lunar orbital rendezvous mission.

CONFIGURATION ANALYSIS

Subsystem Description

The guidance and navigation subsystem is composed of equipment to determine the actual position of the spacecraft at any given time with respect to a specified frame of reference in space; to provide the information and/or control necessary, so that deviations from an intended trajectory can be recognized and corrective maneuvers made; and to serve as an altitude reference for the spacecraft. A description of the functions of the various units is given below.

The inertial measurement unit (IMU) is a three-gimbal, stabilized platform, using three inertial reference integrating gyroscopes (IRIG). It is the function of the IMU to provide a space-stabilized coordinate system and measurements of accelerations. It may be used in the initial earth launch guidance system. It will be used for rendezvous guidance, orientation and guidance for injection into the earth-lunar orbit, mid-course velocity correction measurements, guidance for lunar landing and lunar take-off, and guidance for the earth entry.

The power and servo assembly (PSA) contains such items as platform servo amplifiers, accelerometer electronics, gimbal-mounted preamplifiers, power supplier, etc. It supplies a-c and d-c power for the IMU, as well as for other equipment of the guidance and navigation subsystem.

[redacted]

[redacted]

The Apollo guidance computer (AGC) is designed along the lines of a general purpose computer and is essentially a parallel word transfer machine. Its function is to supply IMU angle pickoffs as whole numbers, accelerometer readouts as increments of velocity, sextant angle readouts, sextant drive, information for pilot display, clock signals for the guidance system, GSE and signals between computer and spacecraft, such as radar altimeter, autopilot, communications, and telemetry. The computer is programmed to give specific trajectories calculated for the normal spacecraft flight path, as well as a succession of trajectories applicable to respective abort modes of operation. The AGC functions in coordination with the IMU to give the precision roll commands necessary to assure survival during entry maneuvers.

The scanning telescope (SCT) is designed to perform low orbit landmark tracking functions. Because of its wide field of view, it also acts as a finder for the sextant. It may be used for platform alinement or in an emergency as a backup for the sextant.

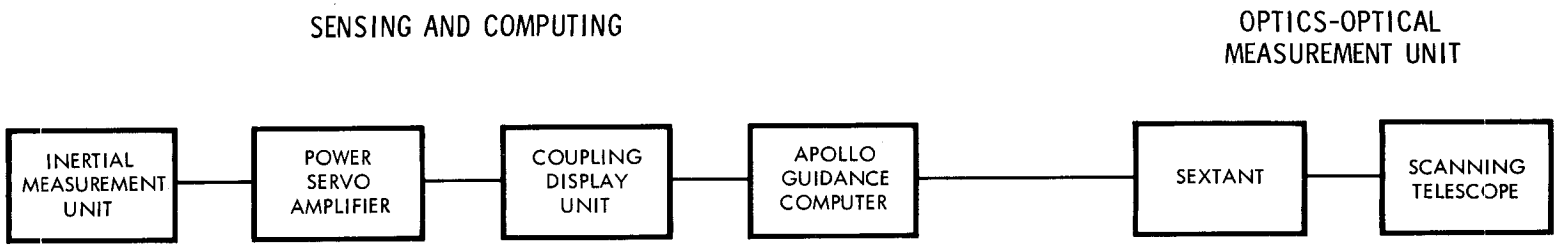
The sextant (SXT) consists of an optical and a controller assembly in which articulating mirrors rotate the sight lines about drive axes. The function of the sextant is to provide primary celestial angle data used for determination of the spacecraft orbit and orientation of the inertial platform.

The coupling display unit (CDU) communicates attitude information to and from the computer, IMU, and the crew. It operates in three modes: IMU coarse aline, IMU fine aline, and attitude measurement. It also supplies attitude error signals to the stabilization and control system in two modes: body axis attitude errors are provided for all mission phases entry; roll and yaw errors about the wind axes are provided during the entry phases.

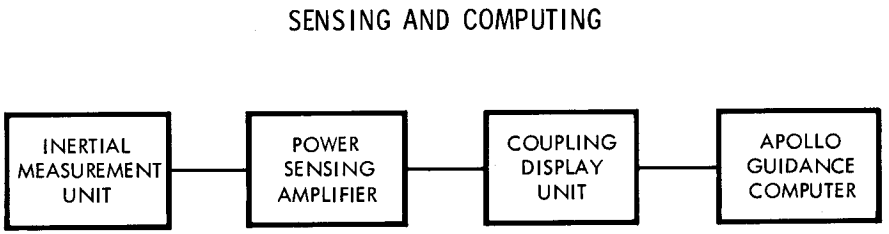
Results of Analysis

Functional diagrams for guidance and navigation and stabilization and control subsystems, showing configurations for navigational sightings, attitude maneuvering, IMU alinements and attitude hold, have been constructed for a 336.41 hour lunar orbital rendezvous mission time line. These data will be used in future studies of integrated systems.

The diagram shown in Figure 34 depicts the reliability mission success configuration for the guidance and navigation subsystem and presents the estimated reliabilities based on Minuteman parts. For mission success these components are in series, since the failure of any one of them would require mission abort. Those components whose failures require abort are also described in the section on Abort Criteria.



PHASE	RELIABILITY	TIME
TRANSLUNAR	0.973500	14.907
TRANSEARTH	0.962750	21.040



PHASE	RELIABILITY	TIME
ENTRY	0.999436	0.333

Figure 34. Guidance and Navigation Subsystem Mission Success Reliability Prediction

Table 46 contains a summary of components within the guidance and navigation subsystem, together with their predicted mission times and reliabilities. Column 4 are the reliabilities using the failure rates supplied by Massachusetts Institute of Technology and no spares. No spares were considered for this first set of reliabilities, since a spares list is not presently available. Column 5 contains the apportioned reliabilities based on the lunar orbital rendezvous mission with the spares noted. Column 6 shows the apportioned reliabilities and spares that would be needed if Minuteman parts are used within the subsystem. The Polaris failure rates were obtained from Massachusetts Institute of Technology and are consistent with those rates being experienced on Mark II Polaris equipment. This analysis indicates the additional spares that are being required within the guidance and navigation subsystem because of the use of non-Minuteman parts. The weight differential indicated is not the total weight differential, since additional spares would be required in the other electronic subsystems noted to meet the overall requirement. For comparison, Column 7 contains the apportioned reliabilities which were assigned for the direct-landing mission. The major reason for this decrease in requirements is that backups were considered in the stabilization and control subsystem.

Planned Activities

Massachusetts Institute of Technology has been requested to submit to NAA/S&ID more refined component reliability estimates, operating times, and levels at which sparing can take place. Based on these modified data, the system analysis will be refined to more accurately establish spares requirements and to establish a firm reliability apportionment.

Table 46 contains a summary of the components within the guidance and navigation subsystem, together with their apportioned and predicted mission reliabilities. No redundancy was considered for the direct mission, and the first lunar orbital rendezvous mission since a list of spares was proposed has not yet been generated. The failure rates used for the direct mission apportionment Minuteman values, comparable to those used in the communications and data subsystem; those used in the first lunar orbital rendezvous mission analysis are based upon the use of Polaris parts as presently proposed by Massachusetts Institute of Technology. In the second lunar orbital rendezvous mission analysis, the spares list was changed from that presently proposed by the addition of a spare coupling display unit. The effect of this additional spare was a decrease in the guidance and navigation subsystem failure rate from 295 to 115 per 10,000 missions. In the third lunar orbital rendezvous mission analysis, the reliabilities are based upon the use of Minuteman parts throughout the entire Apollo electronics and removal of the spare coupling display unit previously considered. The results (42 guidance and navigation subsystem failures per 10,000 missions) indicate the validity of this rationale.

Table 46. Guidance and Navigation Subsystem Mission Reliability

Equipment	Direct Mission	Lunar Orbital Rendezvous Mission								
		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Minuteman	Spares	
	Minuteman		Number	Weight (lb)		Number	Weight		Number	Weight (lb)
Inertial measuring unit	0.998976	0.99400			0.99675			0.99787		
Power servo assembly	0.999925	0.99674			0.99919			0.99991		
Coupling display unit	0.999742	0.99545			0.99985	1	6	0.99973		
Apollo guidance computer	0.999739	0.98786			0.99487			0.99990		
Sextant	0.999845	0.99740			0.99824			0.99991		
Scanning telescope	0.999909	0.99871			0.99970			0.99847		
Total	0.998136	0.97049			0.98864	1	6	0.99579		

STABILIZATION AND CONTROL SUBSYSTEM

During this quarter, the stabilization and control subsystem was analyzed to evaluate its effect on total mission reliability and to establish a spares list and a modified reliability apportionment based on the LOR mission. The analyses indicate that additional spares are necessary to meet the reliability requirement. Based on more detailed analyses and a better definition of possible on-board spares, the analyses will be continued during the next quarter to refine the configuration and the apportionment.

CONFIGURATION ANALYSIS

Subsystem Description

The SCS provides spacecraft stabilization and maneuvering capability during coasting periods, navigational sightings, application of ΔV's, and entry periods. The BMAG provides attitude information to the FDAI through the attitude gyro coupler when it is in the SCS attitude control mode. It can also serve as a backup to the IMU for navigation. Attitude maneuvering is effected by the three axis rotational controls through the electronic control assembly to the reaction control systems when the SCS attitude control mode is in use. The translation control allows the crew to move the spacecraft linearly by way of the reaction control system, and it is also used to provide ullage for a large ΔV maneuver. The SCS control panel allows mode selection and dead-band adjustment. The gimbal position actuator is used to control the positioning of the service propulsion system during periods of large ΔV maneuvers. This information is presented to the astronaut by way of the gimbal position indicator.

The ΔV indicator displays the following information:

(1) The amount of ΔV required during a correction

(2) ΔV remaining

(3) Tail-off correction

(4) Velocity in ft/sec

The following ΔV controls are provided:

(1) ΔV reset
(2) Emergency ullage
(3) Engine fire control
(4) Engine cutoff control.

Results of Analysis

Figure 35 shows the predicted subsystem reliabilities for translunar and transearth phases of the LOR mission with the proposed redundancies. The proposed redundancies include spares for the following components: flight director attitude indicator, gimbal position indicator, stabilization and control subsystem mode select panel, ΔV indicator, one representative module of the electronic control assembly, and three-axis rotation control. Operating times include lunar orbit with LEM lunar ascent as the separation point between orbit times.

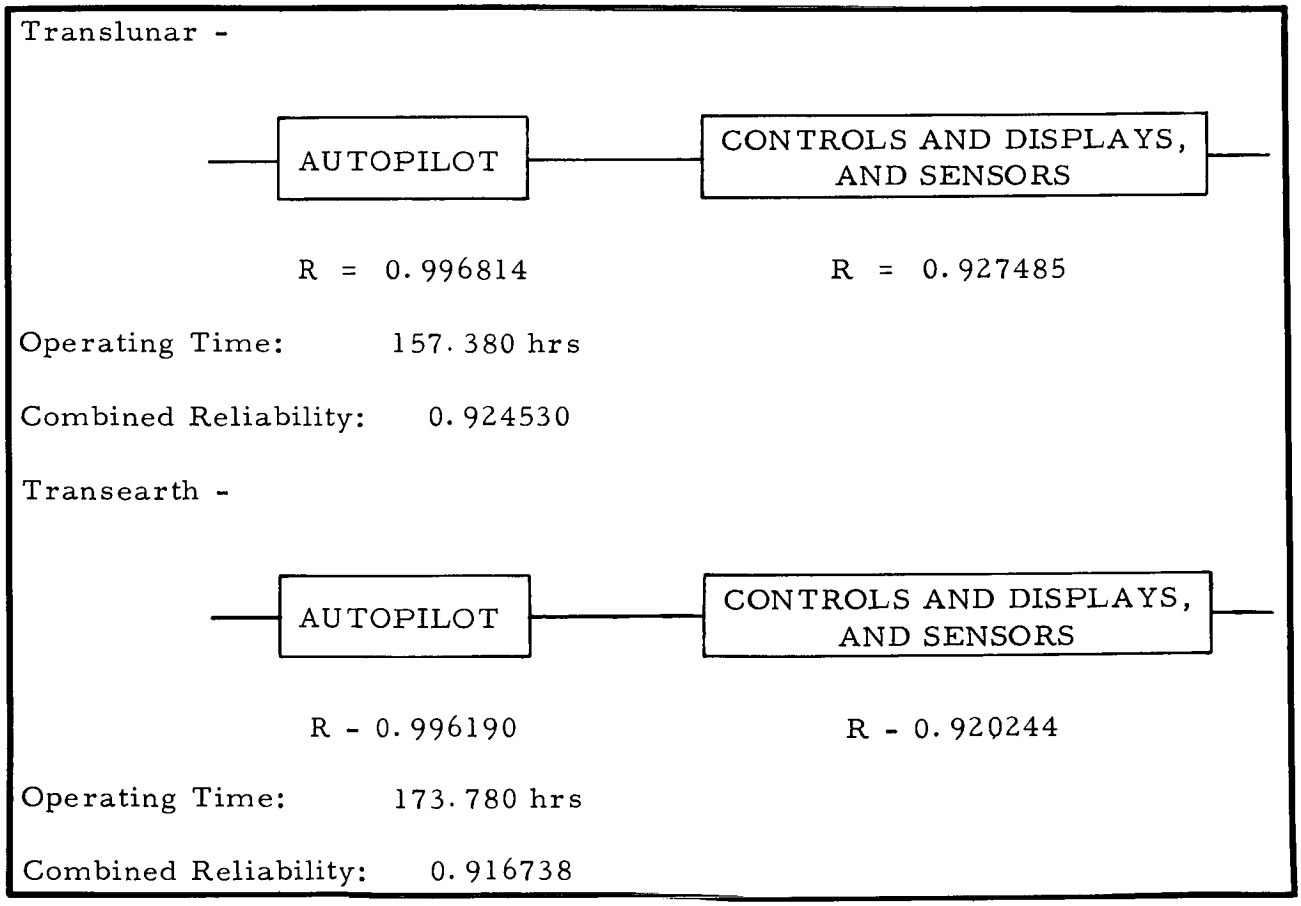


Figure 35. Predicted Reliabilities, Translunar and Transearth Phases LOR Mission

Table 47 shows the operating times and apportioned and predicted mission reliabilities for each of the components within the stabilization and control subsystem. The redundancies and spares considered for the direct mission and the first LOR mission are those presently proposed. The part failure rates used were Minuteman values comparable to those used in the communications and data subsystem. In the second LOR mission analysis, the spares list was changed from that presently proposed. A spare control panel, a spare body mounted attitude gyro, a spare rate gyro, and a spare accelerometer package were added causing a net spares list gain of 35 pounds. The modification of the spares configuration resulted in a decrease in subsystem failure rate from 1200.4 to 102.8 failures per 10,000 missions. In the third LOR mission analysis, the proposed spares list was modified by the addition of one body mounted attitude gyro and one rate gyro package, resulting in a net spares weights increase of 26 pounds. The resultant subsystem failure rate was 248.2 failures per 10,000 missions, a decrease of 145.4 subsystem failures per 10,000 mission from those obtained by the present subsystem configuration; this decrease reflects the additional spares and the use of Minuteman parts throughout the entire Apollo electronics.

TEST PROGRAM

The new qualification test program is intended to accomplish all engineering objectives and replace the qualification and reliability tests previously planned. The objectives of the test program are to qualify the stabilization and control system to specification requirements prior to first manned flight and to provide data that will be integrated with all other developmental field, and flight data for reliability assessment purposes. These tests conducted at black-box component level will qualify the stabilization and control system to maximum specification limits. In addition, off-limit tests to failure will be conducted to verify that design margins are adequate for the more critical failure modes.

The life test phase will consist of system level tests under nominal flight and ground environments (flight environments will be combined in logical combinations where facilities exist) in a sequence simulating an actual lunar and earth return mission. The life test simulations will be completed at least twice on each life system sample for qualification. Considerable reliability assessment information should result from this phase. The present qualification program is based on a no-failure criteria.

The program reorientation should result in a significant cost savings. This is largely because of the reduction in required test sample sizes from 15 equivalent systems to about six equivalent systems. The change in total test effort has not been determined; however, the expanded test of fewer

Table 47. Stabilization and Control Subsystem Mission Reliability

Equipment	Direct Mission	Lunar Orbital Rendezvous Mission								
		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Minuteman	Spares	
	Minuteman		Number	Weight (lb)		Number	Weight		Number	Weight (lb)
Control panel	0.982807	0.983268			0.99932	1	9	0.98420		
Body-mounted attitude gyro	0.99869	0.963797			0.99934	1	12	0.99949	1	12
Rate gyro package	0.99922	0.953607			0.99605	1	9	0.99759	1	9
Electronic control assembly	0.99611	0.99545	1/4	30	0.99834	1/4	30	0.99756	1/4	30
Flight director attitude indicator	0.99755	0.99522	1	12	0.99745	1	12	0.99666	1	12
Delta-V indicator	0.99715	1.0	1	7	0.99968	1	7	0.99978	1	7
Accelerometer package	0.999452	0.983957				1	5	0.99971	1	5
Three-axis rotational control	0.99974	0.998918	1	6	1.0	1	6		1	6
Translational control	0.99988	1.0	1	3	1.0	1	3		1	3
Gimbal position indicator*	0.999995									
Display electronics package*	0.99943									
Total	0.98600	0.87996	4-1/4	58	0.98972	8-1/4	93	0.97517	7-1/4	84

*Deleted from subsystem

systems should result in some savings over the less efficient large sample size testing.

The new test philosophy should provide a well integrated qualification program that will accomplish the system qualification objective prior to first manned flight. Minneapolis-Honeywell has been briefed on the new test philosophy and is currently preparing definite test specifications based on the program changes that will be submit for approval early during the next quarter.

SUBCONTRACTOR MANAGEMENT

A meeting was held at S&ID with representatives of Minneapolis-Honeywell Regulator Company at which a discussion was held on the subject of suppliers submitting justification data on parts that do not appear on the Apollo preferred parts list. Minneapolis-Honeywell was provided with the revised mission success and crew safety electronics subsystems logic diagrams encompassing the failure mode analysis studies. A list of components and estimated failure rates of parts comprising the guidance and control equipments of the spacecraft electronics subsystems was submitted to Minneapolis-Honeywell to aid in their mission success studies. Minneapolis-Honeywell will submit to Apollo Reliability Engineering the latest parts count for the respective equipments in the SCS subsystem as soon as possible.

PLANNED ACTIVITIES

Minneapolis-Honeywell will submit more detailed reliability analyses and a better definition of spares at the module level. To finalize the reliability apportionments that will be issued to the subcontractor, these analyses will be used to refine the analyses indicating that additional spares are necessary. It is expected that providing spares at the lower level will reduce the weight of the spares necessary.

INSTRUMENTATION SUBSYSTEM

During this time period, detailed reliability analyses were performed on the equipments within the instrumentation subsystem, and the effect of these on Apollo spacecraft reliability was evaluated. These analyses indicate that the present configuration is capable of meeting its requirements without spares. During the next quarter, these analyses will be refined to evaluate the effects of subsystem reliability on spacecraft reliability.

CONFIGURATION ANALYSIS

Subsystem Description

In general the instrumentation subsystem supports equipment within other subsystems rather than performing a discrete function.

The central timing equipment provides a time reference for all the equipment within the Apollo spacecraft except for the guidance and navigation subsystem, which has its own timing equipment. It provides time synchronization for the data storage equipment, PCM telemetry, television equipment, in-flight test equipment, and crew displays. It also provides a readout to the crew of Greenwich mean time, time to event, and time after event.

The in-flight test system (IFTS) is a part of this subsystem. This equipment greatly enhances the in-flight maintenance capabilities by detecting malfunctions and troubleshooting them.

The controls and displays section of this subsystem contain all controls and displays that are not directly associated with the major subsystems. These include caution indicators, event time status indicators, entry monitor display, illumination controls, etc.

All sensors that are not a part of other specific subsystems are included under instrumentation. These sensors are used to monitor the various physical variables and environmental conditions of the spacecraft and its subsystems. The derived information is presented to the crew through the displays, monitored by the in-flight test system, and telemetered to earth.

The radiation detection equipment monitors radiation both inside and outside the spacecraft. This data is displayed to the crew and telemetered to the earth. The television and photographic equipment are included in the instrumentation subsystem.

These equipments are used primarily for visual monitoring of the crew and documentation, and are not considered essential to mission success.

All of the antenna equipment used with the communications and data subsystem are included in the instrumentation subsystem except for the VHF/2KMC omni antenna equipment. This equipment includes the C-band antenna, which is used for near earth tracking; the VHF antenna equipment, which is used with the VHF/AM and FM units; the high frequency recovery antenna, and the 2KMC (DSIF) high gain antenna equipment.

Results of Analysis

Table 48 presents mission predicted times and reliabilities for each of the components within the instrumentation subsystem. Column 4 contains the reliabilities that were determined by using Minuteman failure rates and the spares presently planned for inclusion within this system. Column 5 contains the apportioned reliabilities based on the LOR mission using the planned spares and Minuteman failure rates for all subsystems except the guidance and navigation subsystem for which MIT failure rates were used. Column 6 contains the apportioned reliabilities required if Minuteman parts are used for all subsystems. The MIT failure rates are discussed in the section on guidance and navigation. For comparison, Column 2 contains the apportioned reliabilities that were previously assigned for the direct landing mission.

Of the equipment in the instrumentation subsystem only the in-flight test system and the central timing equipment are included in the mission success logic diagrams. Estimated reliabilities of these equipments and their operation times are shown below for near-earth and translunar phases. The IFTS can be considered a partial back up to the communications and data subsystem during near-earth phases while the CTE is considered in a simple series configuration in all phases.

Mission Phase	Equipment	Reliability	Operating Time
Near earth	IFTS	0.999946	4.947
Near earth	CTE	0.9999985	4.947
Combined near-earth reliability =		0.999944	
Translunar	CTE	0.999977	76.28

TEST PROGRAMS

Units Other Than Antennas

The general test program is currently being redefined, and revisions are being incorporated into the program in accordance with the reoriented plan. Since suppliers have not been selected for any units, they have not been effected by the reoriented plan to date. However, it can be reported that the decrease in the number of test units will result in a 20 percent savings in hardware utilization. The test methods and levels will decrease the test effort considerably.

Table 48. Instrumentation Subsystem Mission Reliability

Equipment	Direct Mission	Lunar Orbital Rendezvous Mission								
		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Polaris (Guidance and Navigation) Minuteman (Others)	Spares		Minuteman	Spares	
	Minuteman		Number	Weight (lb)		Number	Weight		Number	Weight (lb)
Central timing equipment	0.9999998	1.0			1.0			1.0		
In-flight test system	0.9998	1.0			1.0			0.99988		
Manual test set	0.99992	1.0			1.0			1.0		
Television system	0.99998	1.0			1.0			1.0		
Sensors	0.9989	0.99844			0.99857			0.99895		
C-band antenna	0.999999	1.0			1.0			1.0		
VHF recovery antenna	0.999999	1.0			1.0			1.0		
VHF recovery antenna backup	0.99999	1.0			1.0			1.0		
2 KMC high-gain antenna	0.9999	1.0			1.0			1.0		
HF recovery antenna	0.999999	1.0			1.0			1.0		
Radome	0.999999	1.0			1.0			1.0		
Photographic equipment	0.9977	1.0			1.0			1.0		
Radiation detector	0.9995	1.0			1.0			1.0		
Controls and displays	0.9999	1.0			1.0			1.0		
Total	0.995593	0.99844			0.99857			0.99883		

Five specifications have been revised to reflect the latest changes. Ten procurement specifications will either be reworked or provided with an initial input showing new mission concepts. Under the reorientation program, the previous schedules will be met or improved, and the test plan will generate sufficient data to assess the reliability of the instrumentation subsystem.

Specific Qualification Tests

No qualification tests have been performed during this reporting period.

Antenna Equipments

The latest hardware requirements for the antenna equipments have been defined in accordance with current program direction. Except for the 2KMC high gain antenna equipment, the number of test units for the antenna equipments were reduced from six to three. The over-all test cost of the antenna equipments should be increased slightly because of the increase in testing levels. There should be no appreciable change in the cost because of the change from mission simulation tests to mission life tests.

The status of the supplier test program is as follows:

Antenna	Status
Operational beacon	Out for bid
2KMC high gain	No specification
Recovery	Contract awarded 12-25-62
R&D beacon	Melpar Inc., qualification testing to begin on 12-17-62
R&D telemetry	Transco, development testing phase
R&D VHF omni filter	Rantec.

Qualification testing is being conducted at Melpar Inc. on the R&D beacon antenna equipment. The tests are:

1. Humidity and combined high temperature and vibration
2. Acoustics and combined high temperature and altitude.

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LEM INTEGRATION

INTRODUCTION AND SUMMARY

A reliability analysis was made on the original proposed configurations for LEM (lunar excursion module) transposition and lunar docking. The configurations studied were (1) mechanical, (2) free-fly-around, and (3) free-fly-around with tethered LEM. Additional studies were conducted on modified configurations consisting of (1) a combination mechanical and free-fly-around mode, (2) a revised free-fly-around with tethered LEM, and (3) a probe and drogue concept. The most reliable of the configurations studied for both LEM transposition and lunar docking was the free-fly-around method with the LEM stabilized by the S-IVB vehicle during transposition.

A qualitative reliability study was made to determine the effect of installing a window in the forward hatch cover. The study indicated that addition of the window would upgrade mission reliability.

CONFIGURATION ANALYSIS

LEM Transposition and Lunar Docking

Reliability analyses have been made on three proposed methods of accomplishing LEM transposition and lunar docking. The methods studied were the following:

1. Transposition and docking with the LEM mechanically restrained
2. A free-fly-around spacecraft
   a. LEM stabilized by the S-IVB vehicle
   b. LEM self-stabilized
3. A tethered LEM, which first involves capture by the spacecraft.

Figures 36 and 37 show reliability logic diagrams of these configurations for the LEM transposition and lunar docking operations. The reliability values used in the calculations were current apportioned values.

		RELIABILITY	FAILURE RATE PER 10^4 MISSIONS
LEM TRANSPOSITION	I. MECHANICAL MODE: TYPICAL COMPONENTS* 0.99970 — TRANSFER ARM 0.99920 — ROTARY JOINT 0.99962 — SHOCK ATTENUATION 0.999972	0.9991	> 9
	II. a. FREE FLY-AROUND, LEM S-IVB STABILIZED: TYPICAL COMPONENTS* 0.99970 — SPACECRAFT STABILIZATION AND CONTROL 0.999970 — SHOCK ATTENUATION 0.999972	0.99965	> 3.5
	II. b. FREE FLY-AROUND, LEM SELF STABILIZED: TYPICAL COMPONENTS* 0.99970 — SPACECRAFT STABILIZATION AND CONTROL 0.999970 — SHOCK ATTENUATION 0.999972 — LEM STABILITY	= 0.99965	> 3.5
	III. FLY-AROUND, LEM TETHERED: TYPICAL COMPONENTS* 0.99970 — SPACECRAFT STABILIZATION AND CONTROL 0.999978 — CAPTURE AND TETHER — CABLE, REEL MOTOR 0.99985 — SHOCK ATTENUATION 0.999972	0.9995	> 5
	*SEE FIGURE 2		
LUNAR DOCKING	I. MECHANICAL MODE: TYPICAL COMPONENTS * 0.99961 — TRANSFER ARM 0.99980 — ROTARY JOINT 0.99962 — SHOCK ATTENUATION 0.999972	0.9990	> 10
	II. FREE FLY-AROUND: TYPICAL COMPONENTS* 0.99961 — SHOCK ATTENUATION 0.999972	0.9996	> 4
	III. TETHERED LEM: TYPICAL COMPONENTS* 0.99961 — CAPTURE AND TETHER — CABLE, REEL MOTOR 0.99985 — SHOCK ATTENUATION 0.999972	0.9994	> 6
	*SEE FIGURE 3		

Figure 36. Logic Diagram, LEM Transposition and Lunar Docking

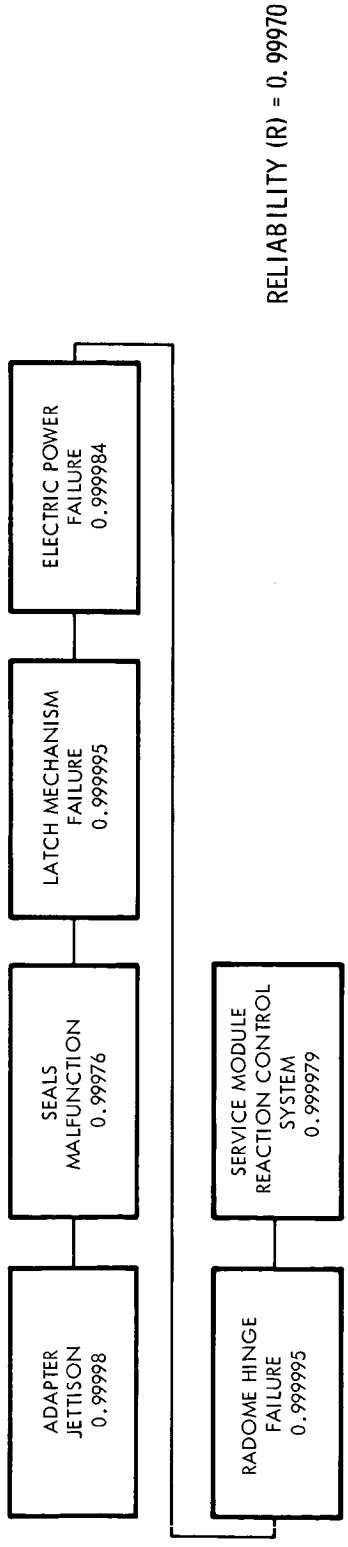
RELIABILITY (R) = 0.99970
ELECTRIC POWER FAILURE 0.999984
LATCH MECHANISM FAILURE 0.999995
SEALS MALFUNCTION 0.99976
ADAPTER JETTISON 0.99998
SERVICE MODULE REACTION CONTROL SYSTEM 0.999979
RADOME HINGE FAILURE 0.999995

Figure 37. Logic Diagram, Typical Components for LEM Transposition

It has been determined from this analysis that all configurations except for the mechanical meet the required reliability. Table 49 presents a summary of the results and a comparison of these with the requirements.

Table 49. LEM Transposition and Lunar Docking

Mission Phase	Configuration	Predicted Failure Rates Per 10^4 Missions
LEM transposition	1. Mechanical	9
	2. Free-fly-around	
	a. LEM stabilized by the S-1VB booster	3.5
	b. LEM self-stabilized	3.5
	3. Free-fly-around, tethered LEM	5
	Required Reliability 0.9992	8
Lunar docking	1. Mechanical	10
	2. Free flying	4
	3. Free flying, tethered LEM	6
	Required Reliability 0.9988	12

Proposed methods of improving the overall reliability include (1) allowing depressurization of the command module, so that man can act as an alternate mode in transpositioning the LEM, and (2) eliminating the requirement for hard-docking at rendezvous. Eliminating the requirement for hard-docking would mean that the crew transfer would be effected in the free-space mode. While specific reliability numbers have not yet been calculated, it is expected that some improvement would result if these proposed methods prove to be feasible.

Ground rules for this study, as set down by the requesting design group, were as follows: (1) no pressurized command module, (2) no command module air lock, (3) use of command module environmental control system for LEM initial pressurization, and (4) no LEM air lock as such, i.e., no exposure of man to space environments during transpositions. Reliability allocation was taken from SID 62-1121, Table D-3, and apportioned as follows:

$$0.998 = R_{Transposition} \text{ and } \cdot R_{Docking} = (.9992)(.9988).$$

The S-IVB vehicle was assumed to be stable, and no probability because of this factor was used. Any factor whereby malfunction would impair the probability of successful hard-dock and necessitate reliance on an emergency mode was considered to have series effect on reliability. The factors common to all transposition modes studied were the following:

1. Adapter jettison failure
2. Malfunction or damage of seals
3. Hard-dock latching mechanism failure
4. Electrical power failure
5. Radome hinging mechanism malfunction
6. Service module reaction control system motors malfunction.

The mechanical mode was affected also by transfer arm and ring system reliability and by shock attenuation system reliability. The free fly-around mode reliability was affected by spacecraft stabilization and control and shock attenuation system reliabilities when the LEM was S-IVB stabilized and by LEM stability reliability in the self-stabilized LEM concept. The free fly-around LEM, tethered mode, required flying the spacecraft; therefore, spacecraft stabilization and control system reliability; probability of capture and tether of the LEM; reliability of the tether, reel, and motor system; and reliability of the shock attenuation system at hard-dock were involved. Any of the modes that required the spacecraft to be flown introduced hazards of collision and damage to the open-hinged radome.

The lunar docking modes introduced those reliability factors resulting from an operative LEM in addition to those bearing on overall transposition

reliability. Reliability factors considered to be common to all modes of lunar docking were as follows:

1. Spacecraft stabilization and control malfunction
2. LEM stabilization and control malfunction
3. Service module reaction control system malfunction
4. LEM reaction control system malfunction
5. Seals malfunction
6. Radome damage
7. Latch mechanism malfunction
8. Collision
9. Electrical system failure
10. Command module, LEM communications.

As with the transposition modes, the lunar docking modes involved reliability factors unique to each mode in addition to the factors listed above.

Since the highest level of reliability possible is desired, it is felt that if the ground rules were modified to allow depressurization of the command module (man-in-space backup to existing modes), some improvement in reliability levels would accrue to the lunar rendezvous. If the requirement for hard-dock at lunar rendezvous were eliminated to permit man-in-space crew transfer as the normal mode, improvement in reliability levels would result. Hard-dock could be reserved as an emergency mode for lunar crew transfer. Hard-docking is obviously required at LEM transposition because of the need for LEM checkout and rigid attachment during midcourse corrections.

Examination of reliability factors common to all transposition and lunar docking modes indicated that a maximum reliability existed independent of the transposition or docking mode selected. This maximum reliability is the result of the design of spacecraft operative systems such as stabilization and control, electrical power, and reaction control system. Of the modes studied, free-fly-around is the most reliable and most nearly approaches the reliability limit for both transposition and docking. The provision of emergency alternative modes can, in some instances, provide redundancy and up-grade reliability.

The LEM restraining mechanism for the mechanical mode was comprised of two transfer arms trunion mounted at opposite sides of the service module with a ring (horse shoe) at the other extremity that function to support the LEM at suitable hard points. Transposition was accomplished by (1) hinging the radome in the open position and jettisoning the adapter and (2) repositioning the LEM and spacecraft with respect to each other as required for hard-dock. This maneuver is accomplished by suitable translations and rotations using service module reaction control system power.

Both Configurations 2 and 3 required the flying capabilities of the spacecraft after adapter jettison. Stabilization and control system and reaction control system operation were utilized to effect required spacecraft translations rotations. Configuration 2B required the spacecraft to seek the passive self-stabilized LEM then home to hard-dock position; Configuration 3 required attitude for final hard-dock. Each mode required operation of a shock attenuation system (crushable honeycomb attenuator cartridge) to reduce final relative velocity to zero at hard-dock. (The maximum allowable closing velocity is 1.0 fps.) The two modules were secured in hard-dock attitude by a latching system comprised of 12 docking clamps and necessary mechanical linkage located near the command module forward hatch. At the LEM-spacecraft interface, sealing was accomplished by means of double O-rings.

LEM Transposition and Lunar Docking, Mechanical and Free-fly Mode

An additional reliability analyses of a combined mechanical and free-fly mode for LEM transposition and lunar docking has been made. It was determined from the analysis that the reliability of the LEM transposition is 9 failures per 10^4 missions and the lunar docking reliability is 4 failures per 10^4 missions. The mechanical transposition did not meet reliability requirements, but the free-fly lunar docking does. Figures 38 and 39 show reliability logic diagrams of this configuration.

The combined mode is comprised of the mechanical mode in LEM transposition and the free-fly mode in lunar docking. It is seen that the combined mode is the same as the mechanical mode except that reliability is degraded by the probability of successful jettison of the transfer arm and ring assembly.

LEM Transposition and Lunar Docking, Revised Tethered LEM Mode

A reliability analysis has been made of a revised tethered LEM mode for LEM transposition (includes hard-dock) and lunar docking.

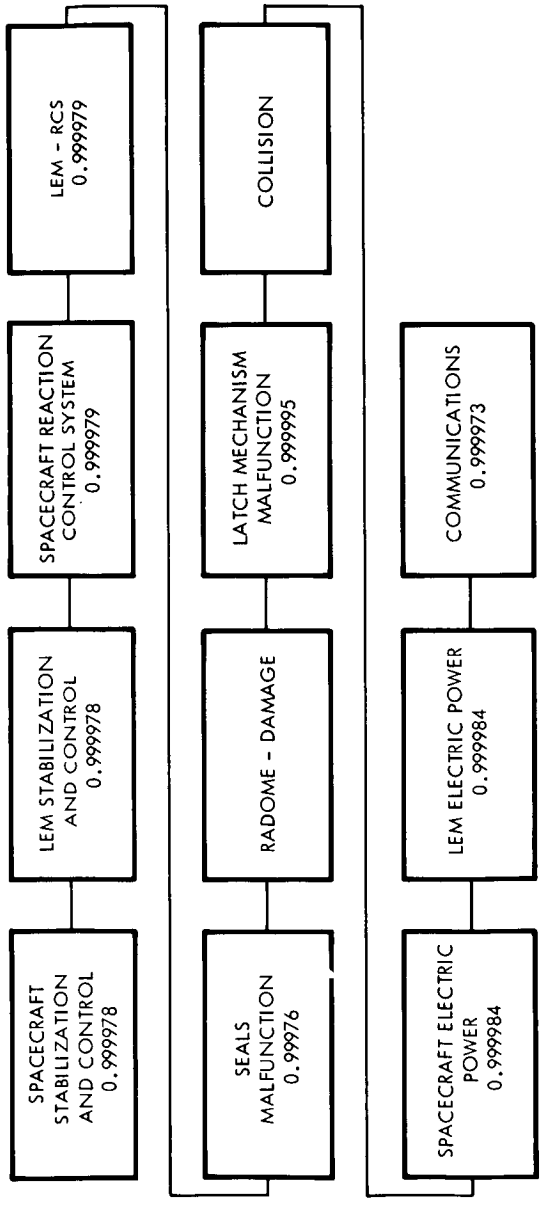

Figure 38. Logic Diagram, Typical Components for Lunar Docking

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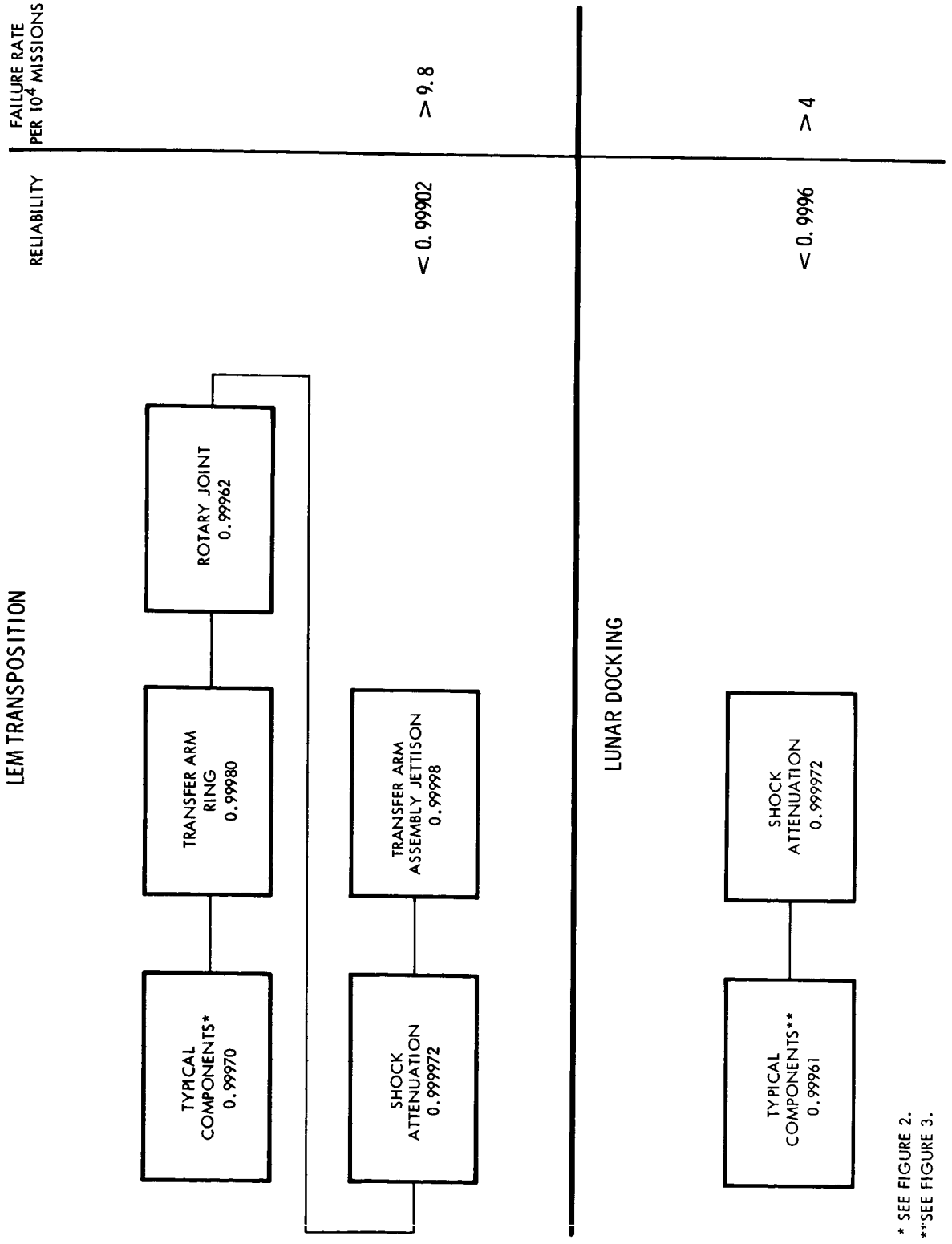

Figure 39. Logic Diagram, Free Flying LEM Transposition and Lunar Docking

As a result of this analysis, the LEM transposition reliability was 0.99925 (7.5 failures per 10^4 missions) and the lunar docking reliability was 0.99920 (8 failures per 10^4 missions). The original tethered LEM mode reliability for LEM transposition was 0.99965 (3.5 failures per 10^4 missions), and for lunar docking was 0.99950 (5 failures per 10^4 missions). Figures 40 and 41 show reliability logic diagrams for this configuration. Both LEM transposition and translunar docking modes satisfy the reliability requirements for mission success.

The tether assembly was comprised of two short arms, rotary joint mounted at opposite sides of the service module to the extremities of which the tether cable was attached. The cable that lay on either side of the service module as far aft as the S-IVB adapter interface, was protected by shrouds during boost.

LEM transposition was accomplished by (1) jettison of the cable shrouds, (2) jettison of the adapter panels, and (3) repositioning the spacecraft as required for hard-dock with the S-IVB stabilized LEM by suitable translations and rotations accomplished with service module reaction control system power.

Lunar locking required the flying capabilities of LEM in addition to that of the spacecraft. Docking was accomplished by (1) flying LEM to the tether cable and effecting hook-up. (2) flying LEM as required to achieve the required relative orientation with the spacecraft and effecting hard-dock.

A shock attenuation system (crushable honeycomb attenuator cartridge) was required to reduce final relative velocity to zero at hard-dock. (The maximum allowable closing velocity is 1.0 fps.) The two modules were secured in hard-dock attitude by a latching system comprised of twelve docking clamps necessary mechanical linkage (located near the command module forward hatch). At the LEM - spacecraft interface, sealing was accomplished by means of double O-rings.

Ground rules for this study were: (1) no pressurized command module, (2) no command module air-lock, (3) use of command module environmental control system for LEM initial pressuring action, and (4) no LEM airlock as such: i.e., no exposure of man to space environment during transposition. Apportioned reliability was taken from SID 62-1121, Table D-3 and redistributed as follows:

$$0.998 = R_T \cdot R_D = (.9992)(.9988)$$

The S-IVB vehicle was assumed to be stable and no probability due to this factor was used. (Note: To achieve the reliability level required for mission success, the S-IVB stability mission acceptable reliability would be

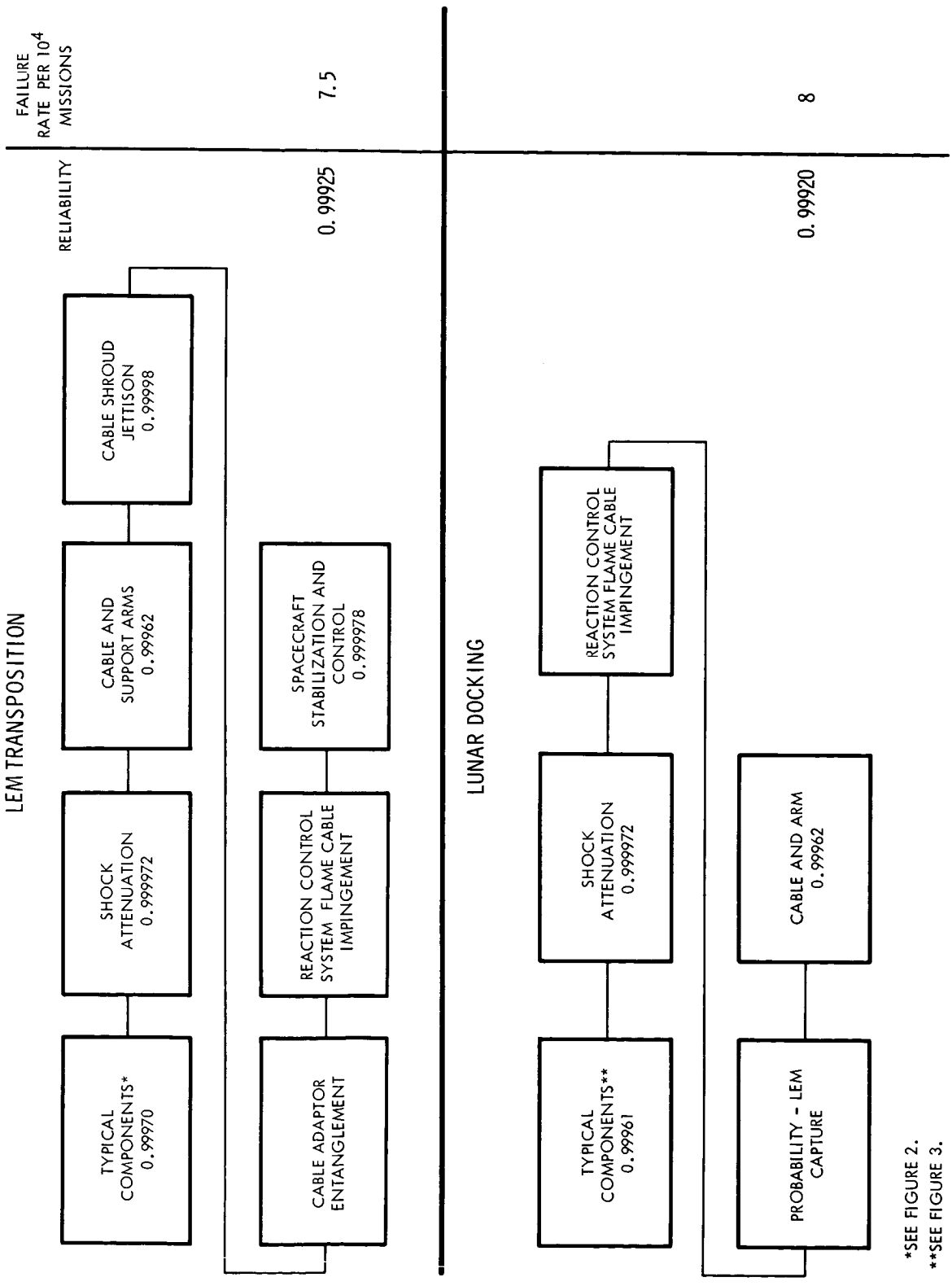

Figure 40. Logic Diagram, Revised Tethered LEM Mode - LEM Transposition and Lunar Docking

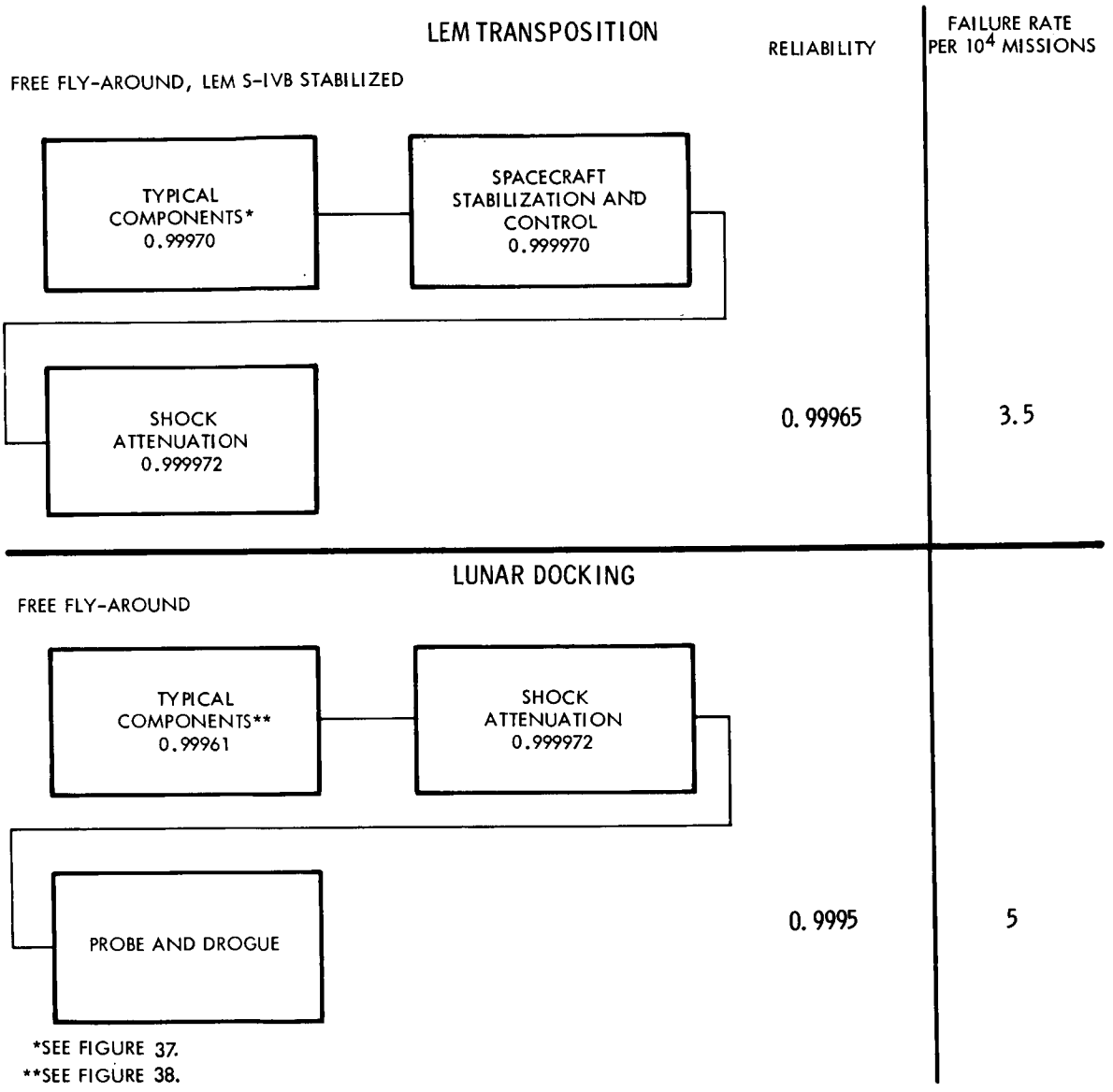

Figure 41. Logic Diagram, Probe and Drogue LEM Transposition and Lunar Docking

0.99945.) Any factor whose malfunction would impair the probability of successful hard-dock and necessitate reliance on an emergency mode was considered to have series effect on reliability.

The tether mode does not incorporate the means for other than holding LEM captive. A previous tether mode provided the means of reeling the cable as an aid to accomplishing hard-dock. Essentially the subject mode required free flying either spacecraft or LEM to hard-dock, the tether may function either to simply hold LEM captive or as means of man-in-space emergency transfer. Additionally, this tether concept introduces the hazards of (1) entanglement with service module reaction control system, (2) possible malfunction of arms and/or rotary joints, and (3) malfunction resulting from entanglement with adapter panels during jettison.

LEM Transposition and Lunar Docking, Probe and Drogue Concept

A reliability analysis was made of the probe and drogue concept of LEM transposition and lunar docking. The concept studied was the free fly-around spacecraft with the LEM stabilized by the S-IVB vehicle. The probe is part of the spacecraft and mates with the drogue contained on the LEM. Figures 40 and 41 show reliability logic diagrams of this configuration.

As a result of this analysis, it has been determined that all evaluated configurations meet the required reliability. A summary of the results and comparison with the requirements are presented in the following table.

Table 50. Probe and Drogue Concept of LEM Transposition and Lunar Docking

Mission Phase	Configuration	Predicted Failure Rates Per 10^4 Missions
LEM Trans-position	Free-fly-around = LEM stabilized by the S-IVB booster	3.5
	Required Reliability 0.9992	8
Lunar Docking	Free flying	4
	Required Reliability 0.9988	12

COMMAND MODULE FORWARD HATCH WINDOW

Description

A qualitative reliability study has been made of advantages and disadvantages that would result from addition of a window in the command module forward hatch cover. (LEM transposition and lunar docking aid) and is shown in Figure 42.

Ground rules for this study were: (1) assume existence of window in command module forward hatch, (2) two way communication exists, (3) no visibility problem due to the crewman's helmet exists, (4) crewman can fly service command from his position in forward hatch, (5) docking windows in command module eliminated and (6) free-flying mode assumed. Some effects of adding a forward hatch window are:

Advantages

1. Docking loads taken through command module and LEM hatch covers would be eliminated.

2. Blind final approach and hard dock of LEM with spacecraft would be eliminated.

3. Better final approach control resulting from direct visual appraisal.

4. Hazard of damage to seals or sealing surfaces at interface would be reduced (spacecraft and LEM both under active control).

5. Direct visual appraisal of sealing malfunction or defect.

6. Direct visual appraisal of any latching malfunction.

7. Preplanning of required repairs in forward hatch and radome areas possible after direct visual assessment.

8. Reduced probability of collision between spacecraft and LEM.

9. Spacecraft crewman could actively aid man-in-space emergency crew transfer from LEM.

10. Appraisal of radome latching and/or hinging mechanism malfunction.

11. Feasibility of seeking and docking with LEM in emergency would be improved.

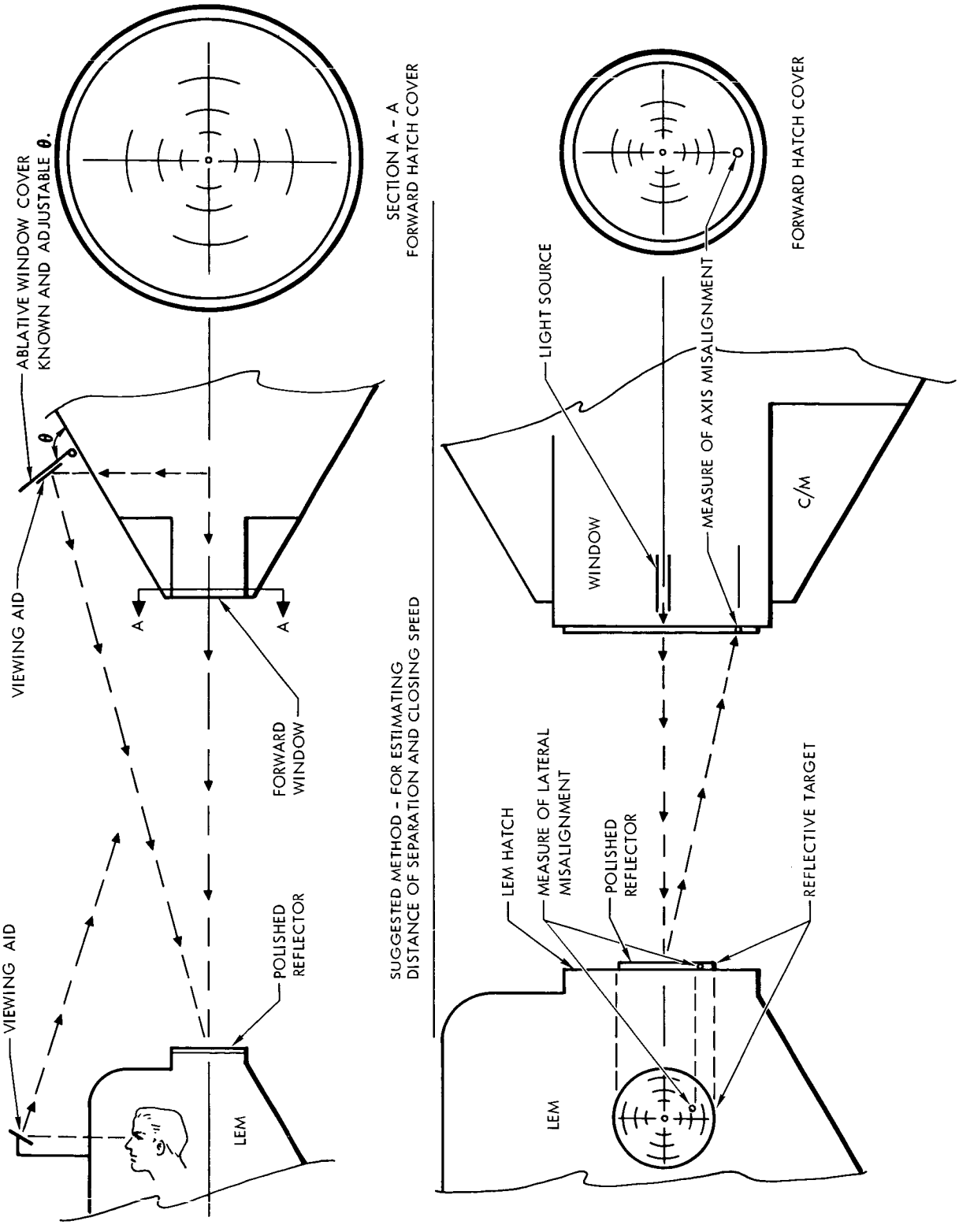

Figure 42. Command Module Forward-Hatch-Cover Window

Disadvantages

1. Shock attenuation axially located is not feasible.

2. Requires all docking operations to be performed by a single command module crewman.

3. Limits locking and sealing means for forward hatch cover (Breech lock).

4. Hazard because of window in hatch cover.

5. Multiple shock attenuators probably required.

Problem Areas

1. Tumbling or passive LEM - emergency.

2. Man-in-space crew transfer - emergency.

3. Alternative - window in LEM hatch cover.

Conclusions

The addition of a window in the command module forward hatch cover would probably upgrade reliability to some extent. Disadvantages resulting from the presence of subject window are not critical. Revision of the present shock attenuator concept would be required, and this would require a subsequent reliability analysis. The means of retaining the hatch cover in position and of compressing the O-ring seals could not include a mechanism which would interfere with viewing.

A window in the hatch cover does offer the advantage of direct viewing of the "target" vehicle during the final moments of approach and hard dock. If the window surface is properly marked and calibrated, it could be utilized as an aid to axis and lateral alignment of LEM with spacecraft (Figure 42).

Use incidental to the prime purpose of the window (for approach and docking) such as (1) direct viewing of interface seals and latching mechanism and (2) inspection of radome mechanisms would tend to upgrade reliability, since preplanning of emergency modes and/or repairs would be possible.

Reliability could also be upgraded by adding simple exterior viewing aids (mirrors), possibly mounted on the inner surface of the docking window

ablative covers. Externally mounted mirrors could also be utilized to eliminate blind zones on the LEM.

PROBLEM AREAS

A survey was conducted and the following problem areas were found to exist.

1. LEM to command module sealing (at hard-dock)
2. Latching at hard-dock due to precise mating required LEM to command module
3. Cold welding in space environment
4. Docking in emergency situation:
   (a) with tumbling LEM
   (b) with passive LEM
5. Man-in-space transfer modes
6. Alternatives to forward hatch window
7. Correctives for visual limitations
   (a) external viewing aids
8. Effect on reliability due to scavenging possibility from LEM (equipment, instruments, etc.)

PLANNED ACTIVITIES

Results of these Reliability studies, together with those from other affected groups, have been included in a report to NASA. Future activities in this area await reaction from NASA.

COMPONENT CRITICALITY

A criticality analysis of the various Apollo spacecraft systems was completed during the previous quarter. The resulting criticality numbers were used to rate systems, subsystems, and components with respect to each other. A high criticality number indicates that there is a high degree of degradation in crew safety caused by the portion of the system being rated.

There are three basic areas where the results of a criticality analysis could be beneficial. The three areas are (1) design improvement, (2) qualification test planning, and (3) instrumentation justification.

The factor that could be used for improving design or deciding on a need for instrumentation is called design criticality. Design criticality is proportional to the number of times that the crew would be lost because of the unreliability of the part. It is computed by multiplying the part unreliability by the partial derivative of the crew safety equation. Since the potential crew loss is decreased proportionately with part reliability improvement, this factor can be used to rank parts for design improvement.

The factor that could be used in the planning of qualification tests is called test criticality. Test criticality is the slope of the crew safety compared to the part reliability curve and is computed by dividing the partial derivative of the crew safety equation by the part reliability. The objective of the qualification test is to verify that part complies with the requirements; however, this must be done in the most economical way. In order to determine the type of qualification test plan that complies with both the reliability requirements and economical limitations, judgment is necessary. Test criticality assists in this judgment by ranking each part with respect to the effect on crew safety of uncertainty in its reliability. When parts have equal reliability uncertainty, the part with the higher test criticality should be emphasized during qualification testing. The resulting test criticality numbers for the various components within each of the Apollo spacecraft system are listed in Table 51.

Instrumentation could be justified partially by the design criticality and partially by abort sensing requirements. The part may be more critical, but if correction is not possible, sensing is not necessary. Design criticality will rank abort and correction sensing needs by numerically rating the contribution of each part to crew loss.

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To aid in the understanding od criticality, the definitions are presented in a mathematical form and applied to several examples.

Test Criticality

$$\text{Test criticality number} = \frac{\partial \text{ Crew safety reliability}}{\partial \text{ Part reliability}}$$

Design Criticality

Design criticality number = (Test criticality) x (1 - Part reliability)

Example 1

$R = P_A P_B P_C$; $P_A = 0.9$, $P_B = 0.8$, $P_C = 0.7$, $= 0.504$

Test Criticality	Ranking
$C_{AT} = \frac{\partial R}{\partial P_A} = P_B P_C = 0.56$	3
$C_{RT} = \frac{\partial R}{\partial P_B} = P_A P_C = 0.63$	2
$C_{CT} = \frac{\partial R}{\partial P_C} = P_A P_B = 0.72$	1

Design Criticality	Ranking
$C_{AD} = \frac{\partial R}{\partial P_A} \times (1 - P_A) = 0.056$	3
$C_{BD} = \frac{\partial R}{\partial P_B} \times (1 - P_B) = 0.124$	2
$C_{CD} = \frac{\partial R}{\partial P_B} \times (1 - P_C) = 0.216$	1

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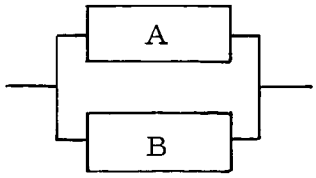
For components in series on the logic diagram, both test and design criticality rankings are the same. The variation in spacecraft crew safety because of component reliability is similar to the variation in number of spacecraft failures due to component reliability. The lowest reliability component has the most criticality.

Example 2



$$R = P_A + P_B - P_A P_B = 0.98$$

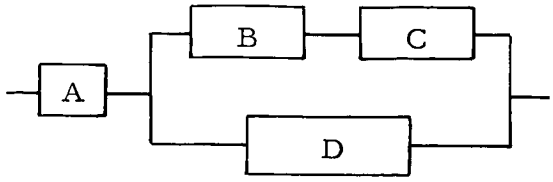
$$P_A = 0.9,\ P_B = 0.8$$

Test Criticality	Ranking
$C_{AT} = \frac{\partial R}{\partial P_A} = 1 - P_B = 0.2$	1
$C_{BT} = \frac{\partial R}{\partial P_B} = 1 - P_A = 0.1$	2

Design Criticality	Ranking
$C_{AD} = \frac{\partial R}{\partial P_A} \times (1 - P_A) = 0.02$	1
$C_{BD} = \frac{\partial R}{\partial P_B} \times (1 - P_B) = 0.02$	1

For parallel components, the highest reliability component has the highest test criticality, because the reliability decrease of the highest reliability component produces more degradation of spacecraft crew safety. Since both components have to fail before the Spacecraft can fail, each parallel component contributes the same number of spacecraft failures. The probability of both components failing on the same mission is independent of sequence. Therefore, either component has the same design criticality regardless of its reliability.

Example 3



$$R = P_A (P_B P_C + P_D - P_B P_C P_D) = 0.846$$

$$P_A = 0.9,\ P_B = 0.99,\ P_C = 0.81,\ P_C = 0.7$$

Test Criticality	Ranking
$C_{AT} = \frac{\partial R}{\partial P_A} = P_B P_C + P_D - P_B P_C P_D = 0.94$	1
$C_{BT} = \partial R/\partial P_B = P_A(P_C - P_C P_D) = 0.2184$	3
$C_{CT} = \partial R/\partial P_C = P_A(P_B - P_B P_D) = 0.2673$	2
$C_{DT} = \partial R/\partial P_D = P_A(1 - P_B P_C) = 0.18$	4

Design Criticality	Ranking
$C_{AD} = \partial R/\partial P_A \times (1 - P_A) = 0.094$	1
$C_{BD} = \partial R/\partial P_B \times (1 - P_B) = 0.002187$	4
$C_{CD} = \partial R/\partial P_C \times (1 - P_C) = 0.050787$	3
$C_{DD} = \partial R/\partial P_D \times (1 - P_D) = 0.054$	2

As in example 1, the lowest reliability series component has the most test or design criticality. The test criticalities of components 3 and 4 follow the patterns of both examples 1 and 2. The most reliable parallel leg has the higher test criticality, and, within the most reliable leg, the lowest reliability series component has the higher test criticality. The

design criticality of components 2, 3, and 4 follow the same example patterns. Each parallel leg has the same design criticality; the component design criticalities within the leg cannot be higher than the design criticality of the entire leg; and the least reliable series component within the leg has the highest design criticality.

Table 51. Apollo Subsystems Criticality Indices

Component	Criticality Indices	Criticality* Class
Launch escape system	0.9991292	(1)
Tower jettison motor	0.050581	(2)
Launch escape motor	0.048333	(2)
Pitch control motor		
Tower	0.99910	(1)
Tower separation	0.999040	(1)
Electronic power system	0.999054	(1)
Fuel cell	0.002519	(3)
Battery and charger	0.009574	(3)
Inverter	0.001368	(3)
A-c bus	0.001997	(3)
D-c bus	0.001997	(3)
One oxygen supply	0.002234	(3)
Hydrogen supply	0.002234	(3)
Service module reaction control system	0.99997	
Thrust chamber	0.0020392	(3)
Injector	0.0020380	(3)
Engine oxidizer valve	0.0020374	(3)
Engine fuel valve	0.0020368	(3)
Propellant tank	0.0020365	(3)
Propellant solenoid valve	0.0020364	(3)
Helium tank	0.0020364	(3)
Flow meter	0.0020360	(3)
Regulator	0.0000041	(3)
Helium solenoid valve	0.0000040	(3)
Filter	0.0000027	(3)
Burst disc	0.0000027	(3)
Check valve	0.0000020	(3)

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Table 51. Apollo Subsystems Criticality Indices (Cont)

Component	Criticality Indices	Criticality* Class
Cap	0.0000009	(3)
Vent valve	0.0000008	(3)
Fill and drain valve	0.0000008	(3)
Relief valve	0.0000002	(3)
Environmental Control System	0.999030	(1)
Suit circuit	0.998760	(1)
Water-glycol	0.998780	(1)
Command module Pressure and Temperature	0.998744	(1)
O_2 Supply	0.998743	(1)
H_2O Circuit	0.998900	(1)
Command module reaction control system	0.999960	(2)
Rocket engine	0.011090	(2)
Squib valve	0.011090	(2)
Positive expulsion tank Oxidizer	0.011085	(2)
Positive expulsion tank Fuel	0.011085	(2)
Helium tank	0.011085	(2)
Reactant fitting group	0.011084	(2)
Reactant tubing group	0.011084	(2)
Helium fitting group	0.011084	(2)
Helium tubing	0.011084	(2)
Burst disc reactant	0.011084	(2)
Filter	0.011084	(2)
Flow meter	0.011084	(2)
Cap	0.000027	(3)
Regulator	0.000006	(3)
Burst disc, relief valve	0.000002	(3)
Filter, relief valve	0.000002	(3)
Check valve	0.000001	(3)
Relief valve	0.0000003	(3)
Vent valve	0.00000001	(3)
Fill valve	0.00000001	(3)

Table 51. Apollo Subsystems Criticality Indices (Cont)

Component	Criticality Indices	Criticality * Class
Heat protection		
Heat shield	0.999050	(1)
Structures	0.999030	(1)
Mechanical	0.999015	(1)
Adapter	0.999010	(1)
Service module structures	0.999002	(1)
Command module structures	0.999003	(1)
Instrumentation		
Central timing unit	0.04774	(2)
Service module service propulsion system		
Helium tank	0.999968	(1)
Gimbal activator, gears and ball screw	0.999962	(1)
Gimbal bearing	0.999950	(1)
Injector, chamber, and nozzle extension	0.999940	(1)
Tank, fuel	0.999935	(1)
Tank oxidizer	0.999935	(1)
Supply line, fuel	0.999900	(1)
Supply line, oxidizer	0.999800	(1)
Filter, fuel	0.999800	(1)
Filter, oxidizer	0.999800	(1)
Orifice, fuel	0.999800	(1)
Orifice, oxidizer	0.999800	(1)
Heat exchanger	0.999473	(1)
Quantity and gaging	0.999027	(1)
Burst disc, relief line	0.009993	(3)
Filter, relief line	0.009993	(3)
Cap	0.009993	(3)
Main valve, fuel	0.006533	(3)
Main valve, oxidizer	0.006533	(3)
Solenoid valve, main valve pilot valve	0.005619	(3)
Main valve actuator valve	0.005615	(3)
Regulator	0.002008	(3)
Solenoid, crew activated	0.002006	(3)
Check valve, helium supply	0.001999	(3)

Table 51. Apollo Subsystems Criticality Indices (Cont)

Component	Criticality Indices	Criticality * Class
Service module service propulsion system (Cont)		
Gimbal activator; motor, clutch, and gears	0.001999	(3)
Check valve, main valve actuator vent line	0.001997	(3)
Cap	0.001000	(3)
Fill and drain valve, fuel	0.000999	(3)
Relief valve, fuel	0.000105	(3)
Relief valve, oxidizer	0.000105	(3)
Fill and drain valve - oxidizer	0.000999	(3)
Helium fill and drain valve	0.000100	(3)
Stabilization and control system		
System less ATT display	0.999987	(1)
ATT display	0.020139	(2)
Electronic control assy.	0.947096	(1)
Rate gyro package	0.887310	(1)
Three-axis rotational control	0.047278	(2)
Translational control	0.047278	(2)
Body mounted altitude gyro	0.020336	(2)
Flight direction attitude indicator	0.0843839	(2)
Gimbal position indicator	0.0811192	(2)
Stabilization control panel	0.963403	(2)
Delta V display	0.0086560	(3)
Accelerometer package	0.0086535	(3)
Navigation and guidance system		
Sextant	0.858461	(1)
Inertial measuring unit	0.0814185	(2)
Coupling display unit	0.080984	(2)
Apollo guidance computer	0.0812244	(2)
Power and servo assembly	0.0808415	(2)
Scanning telescope	0.080827	(2)

Table 51. Apollo Subsystems Criticality Indices (Cont)

Component	Criticality Indices	Criticality* Class
Communications and data system		
VHF beacon recovery	0.1827×10^{-6}	(1)
HF recovery antenna recovery	0.086273×10^{-6}	(1)
HF tranceiver recovery	0.086327×10^{-6}	(1)
CBA, C band antenna	0.0213534	(2)
CBX, C band transponder	0.0215436	(2)
VHF, OA, omni antenna	0.0215136	(2)
Audio center	0.0215629	(2)
VHF, AM transceiver	0.0215596	(2)
DSIF and PA transponder and power amp	0.0479028	(2)
DISH, antenna	0.0054972	(2)
VHF beacon antenna recovery	1.35186×10^{-10}	(3)
Auxiliary VHF beacon antenna recovery	0.036536×10^{-10}	(3)

*Criticality classes are listed according to criticality index ranges as follows:

(1) 0.999987 to 0.858461

(2) 0.0843829 to 0.011084

(3) 0.009574 to 0.036536×10^{-10}.

[redacted]

RELIABILITY PROGRAM IMPLEMENTATION

REORIENTED QUALIFICATION TEST PROGRAM

During the last quarter of 1962, the principal test activities were concentrated on cost optimization of the qualification test program. This effort resulted in what is now termed the "Reoriented Qualification Test Program," which is described in detail in SID 62-1405. The basic intent in the effort was to design a minimum test program consistent with criticality of the application and the high reliability and crew safety objectives of the mission. This approach resulted in a test plan which is formulated to yield design and performance margin data rather than statistically rigorous reliability assessment information. A summary of the salient features of this program is presented in the following paragraphs.

Objectives

Objectives of the "Reoriented Qualification Test Program" are as follows:

1. To employ development test results from fixed-design hardware toward qualification

2. To allow qualification by similarity to Apollo criteria or better

3. To clearly define the scope of the test program, tailored to the needs of each manufactured or procured item

4. To select the minimum test criteria (number of articles, duration of tests, test environments, etc.) on the basis of application considerations, such as the state of the art, and the criticality of the item toward mission success and crew survival

5. To utilize test articles in more than one phase or part of the test plan (when no detrimental stresses are introduced in prior tests), and/or to refurbish test articles when feasible

Advantages

In addition to the optimizing effects afforded by this program, it offers several other advantages. A well defined scope of testing assures uniform programming by subcontractors and suppliers and strengthens S&ID

[redacted]

control over test activities and the resulting data. Using prequalification test results (development data) toward fulfilling qualification requirements reduces the potential delays and expenditures associated with inadequately developed hardware being subjected to qualification tests. Selecting test criteria on the basis of criticality and in relation to the state of the art assures a meaningful and justifiable test plan.

Planning Ground Rules

Supplementing those guidelines that are intuitive from the foregoing objectives, the following ground rules were employed in definitizing the qualification test plan:

1. Zero failure criteria govern; contingencies are not incorporated in the plan.

2. Test planning is predicated upon engineering rational in lieu of statistical conclusions.

3. Qualification is construed as demonstration of manned lunar mission capabilities, within specification requirements, and repeatability of functional and quality characteristics.

4. Reliability demonstration is not a prerequisite for qualification; however, assessment data requirements are met.

5. As an objective, qualification is to be accomplished by the first manned flight.

6. Engineering information and item criticality are evaluated and optimized against cost reductions.

7. Tests are conducted on the most practical level of assembly.

8. Tests conducted on the next higher level of assembly do not duplicate those conducted on lower assemblies.

9. Tests on their sub-tier supplier's components and parts are to be delineated by major subcontractors, subject to S&ID approval, and are to be limited to the minimum testing that will assure success in later tests.

Test Plan Description

The test plan consists of three basic categories applied in two phases. Phase A is comprised of the first two categories, environmental and

off-limit tests; Phase B is the mission-simulation life test.

Phase A-Design Proof Tests

Phase A consists of two parts, The total phase is arranged to confirm design integrity and application suitability of black boxes and practical lower levels of assembly.

Part One, Environmental Tests. The environmental tests, which form Part 1 of this testing phase, will be run at peak levels of expected mission environments. Test articles will be subjected to single environments, sequentially applied, with exposure times commensurate with the analogous critical mission phases. In general, MIL-E-5272 test procedures will prevail. Explosion proof tests will be performed in a simulated spacecraft atmosphere. Electromagnetic interference tests will conform to S&ID specification MC999-0002B.

Part Two, Off-Limit Tests. Off-limit tests are designed to verify the design margins and full capabilities of Apollo hardware under emergency conditions. Affected parameters are increased, in an incremental fashion to, and if possible beyond, the specified design safety factors. Failure modes and fail-safe provisions are also verified by this test. Specific parameters varied in these tests are determined by failure mode analyses.

Phase B-Mission-Simulation Life Tests

Mission-simulation life tests are conducted on complete subsystems (environmental control, telecommunications, etc.) to assure suitable performance, compatibility and the absence of interface problems under natural combinations of select critical mission dynamic environments at nominal stress values. The test time is commensurate with each phase of the mission, starting with ground handling and transportation of the spacecraft and carried through to recovery. Availability of test facilities and schedules will establish the specific constraints on the actual simulation and will restrict environmental test combinations to a maximum of two or three per phase. Articles subjected to mission simulation are required to survive the first simulation cycle without a failure. Upon successful completion of the first cycle, the subsystem will be tested through a second mission cycle or until a failure is encountered. Critical and limited life items, and components that require refurbishment will be replaced or restored to a like-new condition, and subjected to another dual mission simulation cycle to verify repeatable performance.

The S&ID qualification test plan is most compatible with Apollo engineering and mission objectives. Unlike a conventional test program patterned after MIL-E-5272 exclusively, the results from various other test

phases are continuous and correlative with those obtained from qualification. Verification of suitable performance and compatibility at all levels of assembly are assured under single environments and simulated mission conditions, with no duplication of testing. A measure of repeatability of functional and quality characteristics is an inherent part of the program because of the multiple articles subjected to test.

Test Plan Implementation and Integration

As in the past, S&ID is employing a totally integrated test program, starting with development tests, proceeding through qualification, and ultimately including actual spacecraft flights. In addition to assuring correlation of the results from one test phase to the next, the continuity of data resulting from integration permits assessment of MTBF (mean time between failures) at various milestones on the project. The importance of this approach cannot be overemphasized on the Apollo development program, for several reasons:

1. Reliability assessment becomes a by-product of the data gathered from many engineering tests and involves no additional hardware.

2. Because of the high reliability objectives, the MTBF apportioned from these objectives cannot be demonstrated at any level of assembly from components through the complete spacecraft. (Program constraints; such as available hardware, test facility capacity, cost, and schedules; preclude demonstration).

Under these restrictions, some lesser MTBF assessment program must be resorted to. The Apollo assessment plan is predicated upon the simple fact that the spacecraft cannot possibly achieve its reliability objective of 96 percent (8400 hours MTBF), if any subsystem within the spacecraft has failed to reach a reliability in excess of 96 percent. Therefore, the MTBF (referred to as demonstration MTBF) associated with the latter is used to define the total time to be accumulated during all phases of testing.

To date, all equipment procurement specifications have been revised or ammended to include the requirements of the reoriented qualification test plan. Newly issued specifications will also conform to the requirements of the plan. Details on specific test plans for subsystems are contained in the appropriate subsystem sections of this report.

A plan of action has been formulated and implemented to incorporate the reoriented qualification test program into the present Apollo General Test Plan (SID 62-109) and to make the latter more comprehensive series of documents. The present Qualification-Reliability Test Plan (SID 62-204)

will be eliminated, and the applicable contents will be included in several volumes of the General Test Plan. It is envisioned that the new General Test Plan will present a detailed picture of the Apollo test program from development through flight testing. The March 1963 revision will reflect these changes.

This revised General Test Plan will consist of the following volumes under the same basic report number:

Title	Volume
General Test Plan Logic	SID 62-109-1
Individual System Tests	SID 62-109-2
Ground Qualification Test Program	SID 62-109-3
Acceptance Tests	SID 62-109-4
Multiple Systems Tests	SID 62-109-5

Volume 1 will define and emphasize test logic, test integration, continuity of data through all test phases, the reliability assessment plan, and will contain synopses of the remaining four volumes.

Volume 2 will detail development testing of individual systems and will include levels and schedules of environments applicable to this phase of testing.

Volume 3 will be a new volume that will encompass the complete ground qualification program for all spacecraft subsystems as well as identifiable GSE subsystems and applicable lower levels of assembly. This will include a comprehensive coverage of environments, time of test, numbers of equipments to be tested, and detailed schedules for such tests.

Volume 4 will also be a new volume delineating the specific acceptance test procedures to be employed at each applicable level of assembly.

Volume 5 will be a revised version of the current SID 62-109-3 and will encompass all combined systems tests, the new qualification test concept, particularly as it applies to qualification by tests on ground spacecraft, and boilerplate and spacecraft flights. The associated ground support equipment will also be covered in this volume.

In addition to the foregoing, applicable sections of SID 62-203, The Reliability Program Plan, will be revised and resubmitted to NASA-MSC for approval. Reliability assessment status will be documented in subsequent issues of This Quarterly Reliability Program Status Report (SID 62-557 series reports) and qualification progress will continue to be documented via SID 62-784 Qualification Status Report.

DATA OPERATIONS

NONCONFORMANCE REPORTING SYSTEM

The Apollo nonconformance reporting and corrective action system has been defined for machine implementation. To meet various output requirements of the users, it is necessary to divide the programming into the following two phases:

Phase I

This phase of programming, scheduled for completion during the next quarter, will provide automated storage and reporting capabilities for the following types of reports:

1. A summary of failures and discrepancies by part, assembly, and vehicle.
2. Material Review Report of nonconforming parts
3. Selected GSE Report
4. Selected Human Factors Report

Phase II

This phase of programming is expected to be complete by July 1963. The concept in programming this phase is to write generalized programs. The following generalized concepts are being developed:

1. Single-line reporting. Selecting many elements of data from magnetic tape (maximum number of characters 132)
2. Multiple-line reporting. Selects all elements of data to satisfy requests
3. Summaries. Control on any elements of data to produce a report summarizing data for a stipulated period of time

QUALIFICATION STATUS LIST

Preparation of the qualification status list for the fourth quarter of 1962 was completed. All subsystems are now included and up-dated to reflect the most recent requirements. The major revision concerns the addition of a criticality classification, which will aid in test monitoring and instrumentation evaluation. Based upon the present volume of items included in the report, no automation has been attempted; however, it is anticipated that future effort may require the use of computer equipment. The total number of items was increased from 92 to 459 in the Qualification Status List that will be submitted in February 1963.

OPERATING TIME DATA

A procedure has been developed for acquisition and handling of operating time on time-sensitive equipment. Tabular outputs are being designed to produce reports that will be used in evaluating the following:

Reliability assessments on assemblies and subsystems

Mean-time between failures

Total operating time versus limited life

Number of system successes and failures

TOTAL RECORDS ACCESS CONTROL (TRAC) SYSTEM

Programming techniques to produce composite reports of TRAC are being applied as other data systems, which feed into TRAC, are developed. Development of this system is pending finalization of the following programs:

Identification/traceability and configuration

SPUR (separate parts usage record)

Operating time

NCR and SQH (nonconformance report and suppliers quality history)

PATH (performance analysis and test histories)

Qualification status

Parts library tape

SUBCONTRACTOR DOCUMENTATION

Redirection of the documentation requirements for Apollo major subcontractors has resulted in a concentrated effort to further define and clarify what is expected from all Apollo suppliers. All major subcontractors will either be visited, or contacted during monthly reliability coordination meetings to assure the receipt of complete and accurate failure data, qualification status lists, test summaries, and other pertinent reliability documentation. Initial meetings with Lockheed Propulsion and AiResearch have been conducted, and meetings with Marquardt, Aerojet, and Rocketdyne are scheduled for the coming quarter.

INTERSERVICE DATA EXCHANGE PROGRAM (IDEP)

Internal usage of IDEP services has expanded to 23 Apollo groups; 323 reports were distributed during the quarter. Contributions to the program included reports from Life Sciences and Space Physics.

PLANNED ACTIVITY

Format development for parameter specifications data inputs is scheduled for the next quarter. The data inputs will consist of parametric tolerance limits and will be maintained for comparison with incoming test results. Detailed output requirements for the basic data system will also be defined.

RELIABILITY EDUCATION

Educational courses and material presented during this report period are shown in Table 52.

Table 52. Reliability Education Courses

Course Title	No. of Classes Given	Average Attendance	Total Hours
Fundamentals of Reliability Mathematics	13	10	130
Design Analysis Techniques	13	15	293
Apollo Design Review Procedures	1	25	25
Design and Analysis of Experiments	24	5	360
Total	51		808

The Apollo Reliability Program Plan and the Fundamentals of Reliability Mathematics reports are being scripted for closed-circuit television production. These are the first of a number of programs scheduled that will be utilized for in-house and supplier education.

COMPONENT TECHNOLOGY

RELIABILITY PARTS MANUAL

Significant accomplishments for this quarter include completion of Volume I of the Reliability Parts Manual, "Failure Rate and Application Data". Component part failure rates, failure modes, and part application notes are contained in the manual. This manual is intended to be utilized as an aid in the prediction of reliability levels.

STUDIES

Evaluation and investigation of components scheduled for application to the Apollo spacecraft continued this quarter. Several reports and letters were prepared and issued including the following:

1. Relays. Relays being considered for Apollo usage were evaluated. Available reliability information and failure rates for each type were reported.

2. Threaded inserts, thin wall. Thin-wall threaded inserts being considered for use in Project Apollo were investigated. Electroplated samples of this part type indicated evidence of cracks. These defects were documented by enlarged photographs. The probable cause of the problem was hydrogen embrittlement from the electroplating process. The Design Requirements Group, who were engaged in the preparation of a specification for this type of insert, were apprised of these findings. A solid-film lubricant shall be specified in lieu of any electro-deposited plating system through eliminating this potential problem area.

3. Pneumatic fittings. Twenty-two types of pneumatic fittings were evaluated. Available application and capability information is indicated for each type. Three specific fittings were selected for further study and possible testing.

4. Incandescent lamps. Selected incandescent lamps were evaluated. The 6-volt type was selected as the most desirable for Apollo usage.

5. Power control unit. Thirty-one electronic parts utilized in the Westinghouse power control unit were evaluated. It was determined from available data that three capacitors types would have a high failure rate. It was recommended that these three parts be replaced.

6. Cavity amplifier. A cavity amplifier manufactured by Resdel was evaluated. It was noted that problems existed during high-temperature operating conditions. The manufacturer is currently considering a revision to the amplifier in order to resolve the problem.

7. GSE components. Thirty-seven GSE commercial-type parts were evaluated. Available reliability information was reported. Parts that have been superseded by improved versions were noted.

8. Computer diode. The 427M general purpose computer diode was evaluated. Comprehensive reliability information was developed and reported.

SPECIFICATION REVIEW

Specifications were reviewed for the following components:

1. Relays. A proposed relay specification was reviewed. It was found that the specification would not ensure procurement of parts that would satisfy requirements of the Apollo project. Because time was of the essence, Component Technology generated a new document that would provide adequate specification coverage. Apollo Design concurred with the findings and recommendations of Component Technology and submitted the new document to S&ID's Specification Group for coordination and release.

2. Power transfer switches. A specification covering Apollo power transfer switches was reviewed. Revisions and additions were suggested to up-grade the document to required levels.

3. Motor operated switch and over-current relay. A specification covering a motor-operated switch and over-current relay was reviewed. Additional information required to achieve minimum required specification coverage was indicated.

4. High-reliability parts, handling and controls. A first-level or "top" specification, covering handling and controls required for

high reliability parts, was completed and is currently being coordinated.

5. Circuit breakers. A proposed circuit breaker specification was reviewed. It was found that existing circuit breakers would not meet either the application or the specification requirements. Component Technology recommended that this problem be considered by the Parts Improvement Committee.

6. Wire, electrical connectors. Members of Component Technology formulated a team that included representation of Apollo Reliability, Apollo Design, and Apollo Design Requirements. This team provided assistance to the Apollo Electrical Design Group in the generation of wire, feed-through, and electrical connector specifications that were urgently needed in order to meet existing requirements and schedules.

RADIATION EFFECTS STUDIES

The following radiation studies were completed during the past quarter:

1. Radiation effects, components and materials. A summary of a portion of the available irradiation effects data on components and materials has been issued. This includes an analysis of test data from NASA Langley Research Center's Component Evaluation Program. The following reports were issued.

   a. Preliminary Analysis of Proton Radiation Tolerances for Several Transistors.

   b. Hi-Rel Parts Control Study Program-Detailed Investigation-High-Energy Radiation, Hard Vacuum.

2. Apollo radiation requirements. A preliminary analysis of upper bounds on the anticipated Apollo radiation environment for the exterior and interior of the service module and command module for purposes of part specification radiation requirements has been completed. The analysis conducted are delineated in the following reports.

   a. Preliminary Criteria for the Radiation Environment for Project Apollo.

   b. Preliminary Evaluation of the Component Radiation Environment for Project Apollo.

3. Radiation test program proposal. An evaluation-testing program to verify the preliminary results of the radiation effects study program has been proposed. This proposal outlines test methods and procedures, availability of qualified test facilities, parts to be irradiated, and a cost estimate of the program. The report "Hi-Rel Parts Control Study Program-Radiation Test Proposal for Electronic and Electro-Mechanical Parts". has been issued.

RELIABILITY PARTS MANUAL

Volume II of the Reliability Parts Manual (Preferred Parts) is approximately 75 percent complete. It is anticipated that the manual will be completed during the next quarter.

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SUPPLIER SURVEYS

Preaward surveys constituted the major effort during the past quarter. Companies surveyed, their location, and the Apollo product, are summarized in Table 53. During the next quarter, approximately 32 additional preaward surveys will be conducted, primarily in the areas of the in-flight test system, flight simulator, and special products (hardware). Four resurveys are scheduled for potential suppliers who have modified their reliability controls or instituted policies and procedures to establish approved systems.

A forecast of 103 preaward surveys to be performed through December 1962 was made at the inception of the supplier survey program. Actual preaward surveys to date are 110. The forecasted resurveys were 8, and the actual resurveys were 8.

Summarization of all surveys conducted through December 1962 revealed a total of 118 surveys was completed — 110 preaward surveys and 8 resurveys. Copies of the survey summary will be distributed to various departments for their use. Included in the summary is the supplier qualification status and a description of deficiencies observed. This summary will be maintained and up-dated periodically to include post-award survey data as it is obtained. Current revision to the summary will be completed by 15 January 1963 for in-house distribution.

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Table 53. Supplier Surveys

Equipment	Name of Supplier	Location
Ground cooling cart	C. G. Hokanson Ellis & Watts Keco Industries Stratos/Fairchild Budd Electronics Walter Kidde Space Equipment Missimers American Astrosystems Transicold Corp. Auto Controls Laboratories	Los Angeles, California Cincinnati, Ohio Cincinnati, Ohio Bay Shore, New York Long Island, New York Belville, New Jersey Torrance, California Glendale, California So. El Monte, California Montebello, California Los Angeles, California
Central timing equipment	RCA Astro-Electronics Fairchild Camera & Instruments General Electric Advanced Electronics Center Elgin Watch Hughes Aircraft Texas Instruments Astrodata	Highstown, New Jersey Syosset, Long Island, New York Ithaca, New York Rolling Meadows, Illinois Culver City, California Dallas, Texas Anaheim, California
Positive expulsion tanks	Bell Aerosystems B. F. Goodrich Goodyear Tire & Rubber Tapco Group	Buffalo, New York Akron, Ohio Akron, Ohio Cleveland, Ohio
Special Products	Fairchild Precision Metal Products Baker Manley Accessory Products	El Cajon, California So. El Monte, California Whittier, California
Flight simulator	Minneapolis-Honeywell	Duarte, California
Fuel, Oxidizer and helium tanks	Allison Division of General Motors	Indianapolis, Indiana
General capabilities	Air-a-plane General Mills Electronics	Norfolk, Virginia Minneapolis, Minnesota
Hydrogen regulator	B. H. Hadley, Inc.	Pomona, California
Re-surveys	Itt-Industrial Products Molecular Research Systems Engineering Laboratories Eagle Picher Airite Products Hallicrafters	San Fernando, California West Palm Beach, Florida Fort Lauderdale, Florida Joplin, Missouri Los Angeles, California Santa Ana, California

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APPENDIX

SUBCONTRACTOR DOCUMENTS REVIEWED

Document No.	Title
	AEROJET-GENERAL CORPORATION
AGC 14148	Component Specification, GSE, Service Module Rocket Engine Nozzle Closure
AGC 14147	Component Specification, GSE, Service Module Rocket Engine Nozzle Extension, Leak Check Test Set
AGC 14146	Component Specification, GSE, Service Module Rocket Engine Nozzle Plug
AGC 14145	Component Specification, GSE, Service Module Rocket Engine Nozzle Extension Kit
3865-11B	Qualification-Reliability Test Plan
3865-13B	Test Plan
3865-6A	Reliability Program Plan
3865-42A	Simulated High-Altitude Subscale Thrust Chamber, Evaluation Test Plan
3865-42	Simulated High-Altitude Subscale Thrust Chamber, Evaluation Test Plan
3865-3A	Program Plan
3865-10A	Hardware List
3865-13	Test Plan
3865-10-5	Monthly Progress Report
3865-513	Subscale Altitude Performance Estimates
3865-514	Analytical Prediction of Thrust Coefficient for Non-Equilibrium Flow
3865-509	Preliminary Analysis of Nozzle Extension Wall Temperature
3865-2-6	Monthly Weight and Balance Report
3865-10-4	Monthly Progress Report
3865-30-2	Quarterly Reliability Status Report

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Document No.	Title
3865-44	Qualification Status Report
3865-510	Thermal Analysis of the Nozzle Extension Support Ribs
3865-2-5	Monthly Weight and Balance Report
3865-503I	Heat Transfer to N_2O_4 and Aerozine 50 at Low Velocities
3865-502	Analysis of Interstage Vent Area and Shock Wave Location for the 60:1 Nozzle
3865-506	Thermal Analysis of the Nozzle Extension Flange
3865-508	Effect of Gas Ingestion Upon Engine Operation
3865-14-2	Quarterly Progress Report
	AIRESEARCH
SS-1010-R Rev 3	Quality Control Plan
SS-844900	System Check Out Console
SS-1012-R	End Item Test Plan
SS-1014 Rev 2	Design Criteria Specification
SS-845100	Equipment Specification, Pressure Distribution Test Set
SS-1001-R Rev 2	Test Plan
SS-1008-R	Manufacturing Plan
SS-1027-R	Drawing List
SS-1013-R(6)	Monthly Progress Report
SS-1035-R(2)	Quarterly Reliability Status Report
SS-1042-R(7)	Bi-Weekly PERT Report
SS-1019-R(3)	Quarterly Progress Report
SS-1000-R Rev 4	Schematic Diagram, Environmental Control System, System Specification

Document No.	Title
13755	Equipment Specification, Valves, Solenoid, Two-Way
13756	Equipment Specification, Quantity Sensing Probe and Control Package
13768	Equipment Specification, Pressure Switches, Adjustable, Single Throw, Single Pole, Heater System
13707 II	Reliability Test Plan
13703	Program Plan
13707 I	Reliability Plan
13709	Quality Control Plan
13711	End Item Test Plan
13705	Test Plan
13714-1	Monthly Progress Report
13719	Quarterly Reliability Status Report
COLLINS RADIO	
AR 234-2	Component Specification, R&D Radome
AR 232-1	Equipment Specification, R&D Diplexer
AR 233-1	Equipment Specification, R&D Multiplexer
AR 204-1	GSE Design Criteria Specification
AR-113-3	Qualification-Reliability Test Plan
AR-101-8	Monthly Progress Report
AR-238-1	Human Factors Evaluation of Switch Actuators
AR-101-7	Monthly Progress Report

Document No.	Title
	AVCO
RAD-SR-62-102 Rev 1	Manufacturing Plan
RAD-SR-62-252	Monthly Progress Report
RAD-SR-62-249	Monthly Weight and Balance Report
RAD-SR-62-202-6 Rev 1	Drawing List
RAD-SR-62-202-6	Drawing List
RAD-SR-62-202-3	Drawing List
RAD-SR-62-233	Monthly Progress Report
RAD-SR-62-225	Monthly Weight and Balance Report
RAD-SR-62-202-2	Drawing List
RAD-SR-62-208	Quarterly Reliability Status Report
RAD-SR-62-205	Monthly Progress Report
RAD-SR-62-184	Monthly Weight and Balance Report
RAD-SR-62-188	Monthly Progress Report
	BEECH AIRCRAFT
13703	Program Plan
13706A	Manufacturing Plan
13751A	Equipment Specification, Quick Disconnect Valve Couplings
13752	Equipment Specification, Module, Oxygen System Valves
13753	Equipment Specification, Module, Hydrogen System Valves
13754	Equipment Specification, Check Valve Equipment

Document No.	Title
AR-121-3	Quarterly Reliability Status Report
AR-120-3	Quarterly Progress Report
	LOCKHEED AIRCRAFT
588-M-13	Monthly Quality Report
588-P-6	Monthly Progress Report
588-P-5	Monthly Progress Report
588-M-29	Quarterly Quality Control Performance Audit
588-M-31	Quarterly Reliability Status Report
588-M-28	Drawing List
588-M-27	Drawing List
588-Q-2	Quarterly Progress Report
588-M-26	Monthly Quality Report
	MARQUARDT
A-1006A	Quality Control Plan
A-1002A II	Reliability Program Plan
A-1009A	Hardware List
A-1007A	Test Plan
A-1008A	End Item Test Plan
A-1020-5	Monthly Quality Report
A-1011-5	Monthly Progress Report
A-1020-4	Monthly Quality Report
A-1011-4	Monthly Quality Report

Document No.	Title
A-1015-7	Drawing List
A-1019-1	Quarterly Quality Control Summary
A-1020-3	Monthly Quality Report
A-1028	Quarterly Reliability Status Report
A-1025	Qualification Status Report
A-1026	Quarterly Progress Report
A-1015-6	Drawing List
MELPAR	
A1004.03	Qualification-Reliability Test Plan Reliability Program Plan
MINNEAPOLIS-HONEYWELL	
A62-751G(3)	Support Plan
A62-752C(9)	Monthly Progress Report
A62-752C(8)	Quarterly Progress Report
A62-768D(4)	Monthly Quality Report
A62-760B5.3(1)	Boilerplate, Stabilization and Control System, Stability Analysis Launch Escape, Stabilization and Control System Data, GSE
A62-768OD(3)	Monthly Quality Report
A62-760A8(1)	Stabilization and Control System Failure Indication Study
A62-752C(7)	Monthly Progress Report

Document No.	Title
	NORTHROP
59312	Equipment Specification, Fusing Resistor
59319	Equipment Specification, Explosive Bolt, Electrically Initiated, Drogue Disconnect
59311A	Equipment Specification, Cutter, Mechanically Initiated, Reefing Line
59303A	Equipment Specification, Cartridge, Electrically Initiated
2526A	Reliability Demonstration Plan
2531A	Quality Control Plan Quality Control Training Plan
59332	Quality Control Requirements for Suppliers
2523C	Overall Test Plan
59311	Equipment Specification, Cutter, Mechanically Initiated, Reefing Line
59305	Equipment Specification, Switch, Time Delay
59314A	Equipment Specification, Relay, Non-Latching
59302A	Equipment Specification, Baroswitch
59301A	Equipment Specification, Switch, Inertia
NVC/62-3225/734	Drawing List
2620-12	Monthly Failure Summary
2540 II	Quarterly Reliability Status Report
NVC/62-3105/734	Drawing List
PTM-434A	Hardware List

Document No.	Title
2624-10	Monthly Failure Summary
NVC/62-2939/734	Drawing List
NVT/62-519	Monthly Progress Report
2624-5	Monthly Failure Summary
	Qualification Status Chart
2600-III	Quarterly Progress Report
NVC/62-2520/734	Drawing List
PRATT & WHITNEY	
PWA 2114	Development Test Plan
PWA 2115	Reliability Program Plan
PWA 2116	Test Plan
PWA 2059B	Quality Control Plan
PWA 6342D	Equipment Specification, Shipping Container, Fuel Cell Powerplant
PWA 6352A	Equipment Specification, Test Stand, Fuel Cell Powerplant
PWA 4-10	Monthly Progress Report
PWA 20-4	Monthly Quality Report
PWA 24-11	Drawing List
PWA 24-10	Drawing List
PWA 24-9	Drawing List
PWA 4-9	Monthly Progress Report
PWA 20-3	Monthly Quality Report

Document No.	Title
PWA 2105	Quarterly Reliability Status Report
PWA 24-8	Drawing List
PWA 20-2	Quarterly Summary of Quality Control
PWA 21-3	Hardware List
PWA 24-7	Drawing List
PWA 2101	Quarterly Progress Report
PWA 24-6	Drawing List
PWA 24-5	Drawing List
THIOKOL	
A-013A	Reliability Test Plan
A-10A	End Item Test Plan
A-711	Drawing List
A-710	Drawing List
A-204	Monthly Progress Report
A-709	Drawing List
A-502	Quarterly Reliability Status Report
A-708	Drawing List
A-602	Quarterly QC Summary
A-707	Drawing List
A-302	Quarterly Progress Report
A-706	Drawing List

Document No.	Title
MASSACHUSETTS INSTITUTE OF TECHNOLOGY	
C-1199	Apollo Guidance and Navigation Report
PROCUREMENT SPECIFICATIONS CONTRIBUTIONS	
Model C14-032	Antenna Check Out Group Connector, Flexible, Service Propulsion System Feed Lines General Purpose Relays General Usage Electrical Connectors Heat Exchanger, Service Propulsion System Feed Lines Hose Assembly, Flexible Metal, Service Module
Model C14-019	Launch Control Group
Model C14-031	On-Board Recorder Check-Out Unit Performance and Interface Command Receiver Check Out Unit
Model C14-053	R&D Instrumentation Bench Maintenance Equipment
Model C14-112	Radar Transponder and Recovery Beacon System, Quantity Gaging, Reaction Control System
Model C14-028	Telemetry Check Out Unit
Model S14-016	Water-Glycol Supply Unit Wire to Be Used in Environmentally Controlled Areas
MC 183-8000	Trailer, Transport, Aircraft, Apollo Command Module
MC 284-0026	Valve, Relief, Absolute Pressure, Helium
MC 284-0027	Valve, Relief, Pressure
MC 284-0045	Valve, Monomethyl Hydrazine, Solenoid Actuated

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Document No.	Title
MC 284-0050	Ventilating Check Valve, Waste Management System
MC 284-0062	Valve, Relief, Pressure, Helium, Command Module, Reaction Control System
MC 284-0068	Check Valve, Parallel, Helium
MC 363-8000	Supply Unit, Atmosphere, Ground
MC 364-0001 B&C	Heat Shield, Ablative Panels
MC 414-0061	Electrical Connectors, Special Purpose
MC414-0065	Connector, Umbilical, Electrical, Command Module to Service Module
MC414-0067	Umbilical, Electrical, Launch Escape Tower
MC 432-0040	In-Flight Manual Test Set
MC 449-0005	Transducer, Pressure
MC 449-0014	Transducer, Displacement
MC 449-0015	Transducer, Flow
MC 449-0021	Transducer, Temperature
MC 452-0036	Power Transfer Switch
MC 452-0038	Power Switch and Over Current Relay, Motor Operated
MC 453-0005	Cartridges, Electrically Initiated
MC 453-0007	Firing Unit, Electronic, Exploding Bridgewire
MC 453-0008	Initiator, Exploding Bridgewire
MC 453-0009	Initiator, Electrical, Hotwire
MC 455-0014	Control Relay, General Purpose

Document No.	Title
MC 464-0015A	Fuel Cell, Electrical Power Plant
MC 464-0022	Rectifier, 28-V, D-C Power Supply
MC 481-0001A	R&D Telemetry Equipment
MC 481-0002	Antenna System, Recovery
MC 481-0003A	R&D Beacon Antenna
MC 481-0005	Operational Beacon Antenna System
MC 481-0006	C/M Radome
MC 481-0008	High Gain Antenna Equipment
MC 481-0028	R&D VHF Antenna Coupler
MC 481-0029	R&D C-Band Antenna Coupler
MC 481-0030	R&D Telemetry Coupler
MC 496-0005	Motor Generator Set, 400 Cycle
MC 901-0005B	Cryogenic Subsystem
MC 901-0008A	System, Propellant Quantity Gaging, Service Module, Reaction Control System
MC 901-0009A	Rocket Engine, Stabilization and Control System
MC 901-0024	Vacuum Cleaner Assembly, Waste Management System
MC 901-0025	Waste Management Control Unit
MC 901-0027	Bacteria Control Unit, Waste Management System
MC 901-0029	Urine Disposal Lock
MC 901-0030	Waste Management Blower
MC 901-0039	Pre-Prototype, Stabilization and Control System, Bench Maintenance Equipment

Document No.	Title
MC 901-0041A	Stabilization and Control System, Auxiliary GSE
MC 901-0057	Fuel Cell Powerplant Test Stand
MC 901-0063	In-Flight Test System
MC 901-0067 and A	Rocket Engine, Bi-Propellant
MC 901-0068	Back-up Valve, Waste Management System
MC 901-0073	LH_2 Transfer Unit
MC 901-0074	Parachute Subsystem, Pre-Prototype, Bench Maintenance Equipment
MC 901-0075	LO_2 Transfer Unit
MC 901-0080	Acoustic Measurement System
MC 901-0081	Equipment, Signal Conditioning
MC 901-0090	Command Module Television Equipment
MC 901-0101	Helium Booster Unit
MC 901-0107	Bulkhead Feedthrough, Command Module
MC 901-0108	Service Module Rocket Engine Motor Nozzle
MC 901-0109	Service Module Rocket Motor Leak Test Set
MC 901-0110	Service Module Rocket Engine Motor Nozzle Plug
MC 901-0111	Service Module Rocket Motor Nozzle Extension, GSE
MC 901-0112	Vibration Measuring System
MC 901-0113	Leak Detection System
MC 901-0114	Stress Measuring System
MC 999-0002B	Electromagnetic Interference Control

Document No.	Title
MC 999-0019	Cable Assemblies, Special Purpose Electrical
MC 999-0020	Cabinet and Console Enclosures
MC 999-0023	Communications and Data Subsystems
MC 251-0005	Burst Disc Assembly, Monomethyl Hydrazine
MC 449-0020	Transducer, Linear Acceleration
MC 454-0003	Circuit Breakers, Hermetically Sealed
MC 461-0002A	Charger, Battery, Silver Oxide Zinc
MC 461-0003A	Battery, Stabilization and Control, Storage, Zinc Silver Oxide, 28-V, 30-Amp
MC 901-0012A	Stabilization and Control System Subsystem
MC 999-0008A	Documentation Requirements for Project Apollo Supplies (Components) Apollo Stabilization and Control, Service Module Propulsion System, Simulated Rocket Engine Fuel Transfer and Conditioning Unit Switch Assembly, Command Module to Service Module, Separation Indication
SPECIFICATION CONTROL DRAWINGS (SCD's)	
ME 106-0012	Weighing Kit, Electric, For Determination of Static Weight and Center of Gravity
ME 161-0001	Control, Push Pull, Rigid
ME 414-0069	Electrical Connector Assembly, Boiler Plate
ME 415-0003	Socket, Relay, 16 Pin, Microminiature
ME 415-0004	Socket, Relay, 8 Pin, Microminiature
ME 415-0005	Socket, Relay, DPDT

Document No.	Title
ME 415-0006	Socket, Relay, 16 Contacts
ME 415-0007	Socket, Relay, 10 Contacts
ME 415-0008	Socket, Relay, Microminiature, 8 Pin
ME 441-0019	Capacitor, Fixed, Electrolytic
ME 443-0044	Resistor, Fixed, Wirewound, Power
ME 455-0015	Relay, Armature, Subminiature, 8 Pin
ME 455-0016	Relay, Armature, Subminiature, 9 Pin
ME 455-0017	Relay, Armature, Subminiature, 14 Pin
ME 455-0019	Relay, Armature, Magnetic Latching, Microminiature
ME 455-0020	Relay, Armature, Magnetic Latching, Subminiature
ME 455-0021	Relay, Armature, Latch
ME 455-0022	Relay, Armature, DPDT, Microminiature, 8 Pin
ME 455-0024	Relay, Power, 3PST, Sealed with Rectifiers
ME 479-0019	GSE Diode

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TECHNICAL PROPOSALS REVIEWED

Equipment	Company
C-Band Transponder	Aero Geo Astro Hazeltine ACF Electronics Scope Temco Motorola
DSIF Transponder	Motorola
Central Timing Equipment	Electro Mechanical Res. Fairchild Pioneer Central (Bendix) RCA General Electric McDonnell Texas Instruments Motorola Astrodata Elgin ITT Hughes
PCM Telemetry Equipment	Radiation, Inc Texas Instruments, Inc United Electro-Dynamics EPSCO, Inc Applied Electronics & Baldwin-Lima-Hamilton Corp Tele-Dynamics, American Bosch Arma Corp
Positive Expulsion Tanks	Airite Products Bell Aerosystems Beech Aircraft

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Equipment	Company
Regulators (Helium), High-Flow	Hadley Future Craft Calmee Whittaker Vacco Thickol
Regulators (Helium), Low-Flow	Fairchild (Stratos) Hadley Apco Thickol
C/M RCS Squib Valves	Pelmec Futurcraft Pyrodyne Pyroelies Conax
Check Valves	Sargent Apco Langley Firewel
Static Inverter	Westinghouse

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